SEDSAT Final Report

1 Summary

Under this contract the Students for the Exploration and Development of Space Satellite number One (SEDSAT-1 or just SEDSAT) was completed and launched on 24-Oct-98. It achieved nominal orbit and activated its transponders after Delta II separation. Telemetry has indicated the internal systems are normal. However, SEDSAT has not responded to subsequent uplink commands. Telemetry has continued to be received allowing evaluation of bus performance and attempts at anomaly resolution. Anomaly resolution will continue during 1999 subsequent to the end of this contract. This final report documents activities under the contract period.

2 SEDSAT-1 Mission Description and Objectives

The SEDSAT-1 is the NASA/University of Alabama Huntsville Students for the Exploration and Development of Space satellite number 1 (SEDSAT-1). The purpose of the satellite is to advance the space education mission of the university and NASA, train students in a hands-on mission environment, and conduct science and engineering research.

2.1 SEDSAT Mission Objectives

The technical objectives are to demonstrate the implementation of advanced electronics, power systems, communications, and control algorithms; and perform remote sensing experiments; within the context of a student centered program of development and education. The high quality free remote sensing experiments were designed to take pictures of the earth after the satellite is ejected from the Boeing/McDonnell Delta II rocket. All the information is being placed on World Wide Web servers. The satellite (if/when fully operational) will provide services to the international amateur radio community.

The operational sequence for the satellite is:

1. Separation from the Delta II rocket

2. SEDSAT-1 automatically boots and takes a sequence of images as it separates from the Delta II. It immediately begins to attempt ground communication.

3. Once communications is established, the Delta II images are downloaded and the flight software is uploaded.

4. On command, the stowed antennas are deployed and full operational capability is established.

2.1.1 Communications

SEDSAT-1 will operate as a worldwide amateur radio communications link. The American Radio Relay League (ARRL) and the Radio Amateur Satellite Corporation (AMSAT) will provide tracking information and data acquisition/relay.

SEDSAT-1 will also provide a test bed for new forms of integration of scientific communication with the Internet. In the baseline configuration SEDSAT-1 will archive all of its telemetry and downloaded imagery through the World Wide Web. In a planned experiment, SEDSAT-1 will become a full-fledged Internet node accessible in real-time (albeit intermittently) directly through the Internet.

2.1.2 Earth Remote Sensing

The SEASIS imaging system objectives are to measure the planetary cloud cover, lightning distribution, and the diffusion of light by the Earth's atmosphere.

2.1.3 Technology Development

SEDSAT-1 will test new technologies in batteries, power management architectures, and high power in-situ processing of remote sensing data. Data will be collected on the on-orbit performance of the Marshall Space Flight Center supplied NiMh batteries. Power management will be distributed, utilizing local power conversion. The mass memory technology will be evaluated for space performance. Lastly, the control algorithms are new in the context of this application, magnetorquer control of a symmetric satellite.

2.2 Launch Considerations

SEDSAT deployment is controlled entirely by the Delta II. SEDSAT is non-operational during the entire ascent sequence. After the primary payload, Deep Space-1, is deployed the Delta II second stage performs a depletion burn. The second stage is commanded to the deployment attitude and is then commanded to release SEDSAT by activating the Marmon clamp pyro-cutters. The nominal SEDSAT release is at 5000 seconds when the second stage comes into view of the Hawaii tracking station near the end of the first orbit. Separation springs provide the energy to move SEDSAT away from the second stage. As SEDSAT moves away from the second stage the separation switches close and SEDSAT activates.

3 HARDWARE DESCRIPTION

The SEDSAT-1 hardware consists of the satellite structure, internal satellite subassemblies, body mounted GaAs/Ge solar arrays, and an integral Marmon clamp payload interface.

3.1 SEDSAT-1

SEDSAT-1 is a micro-satellite that consists of four major subsystems. These are the structure subsystem, launch vehicle interface subsystem, electronic assemblies and power subsystem, and the experiment subsystem.

3.1.1 SEDSAT-1 Structure

SEDSAT-1 has a design flight weight of 80 pounds. The satellite is a 13.65 X 13.65 X 12 inch near cube. Materials for the structure were bought from the approved list of structural materials and conform to MSFC-SPEC-522B "Design Criteria for Controlling Stress Corrosion Cracking." The materials used are listed in UAH-SEDSAT-0400. Materials selection was with the original intention of flying as a Hitchhiker payload on the space shuttle.

Five sides of the satellite are 1.27 cm (.5") 6061 T651 aluminum. The sixth side, which is the launch vehicle interface, is 7075 T73 aluminum to comply with the expected loads associated with the Marmon clamp. Figure 3.1.1-1 is a cutaway of SEDSAT-1 with the -X face removed showing the internal components. SEDSAT is flown as a Delta II secondary payload. Its position on the Delta is shown in figure 3.1.1-2.

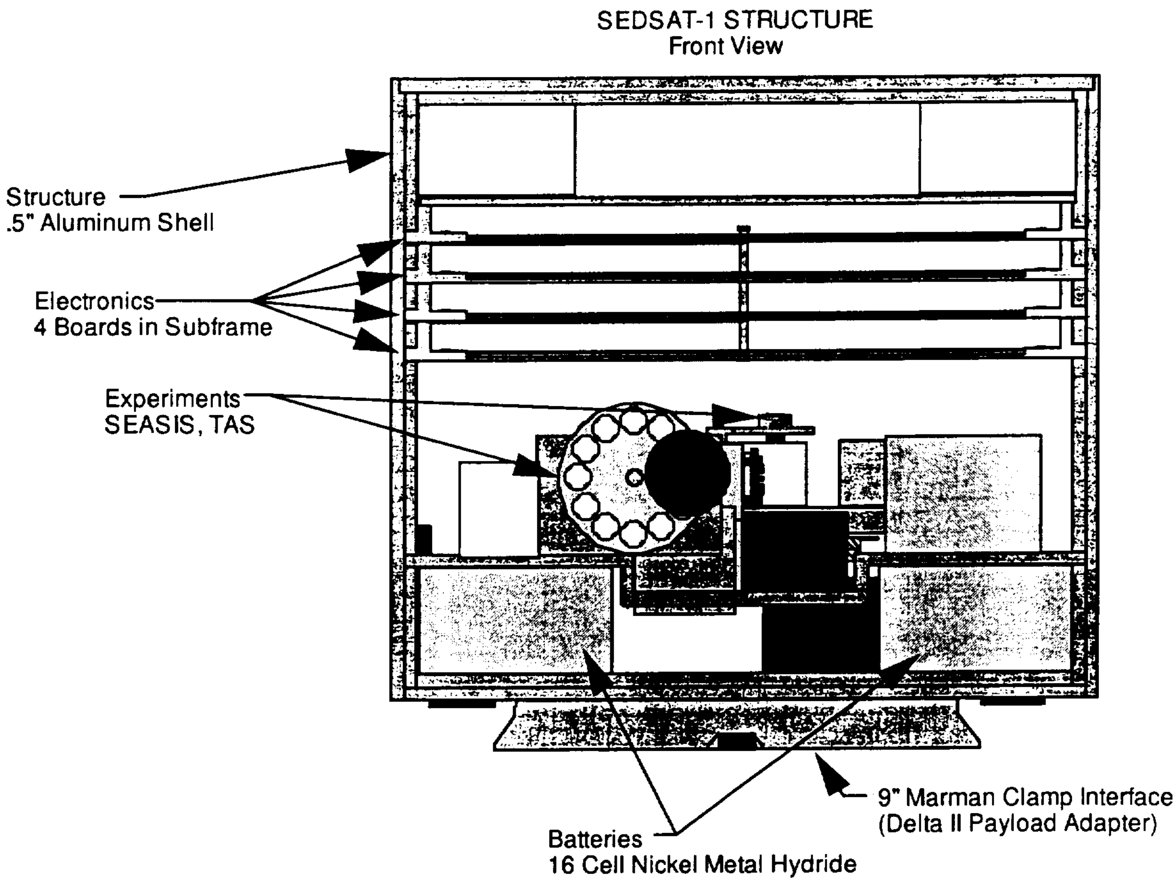


Figure 3.1.1-1 SEDSAT-1 Internal Hardware Placement

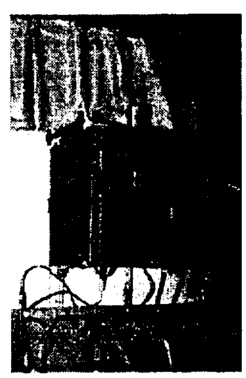


Figure 3.1.1-2 SEDSAT-1 on Delta II before fairing attachment

3.1.2 SEDSAT-1 External Configuration

Five of the six SEDSAT-1 surfaces contain GaAs/Ge solar arrays. They were constructed by TRW and Applied Solar Energy on a structural substrate provided by UAH.

The bottom (sixth side) of the satellite is the ejection system interface, consisting of a 9" Marmon Clamp. This interface was fabricated as an integral part of the bottom of the satellite to improve structural margins and is mechanically compatible with the Delta II Payload Adapter (PA).

The -X face of the satellite has only one protuberance, the SEASIS panoramic imaging camera lens and mount. Figure 3.1.2-1 shows external views of SEDSAT-1 with dimensions.

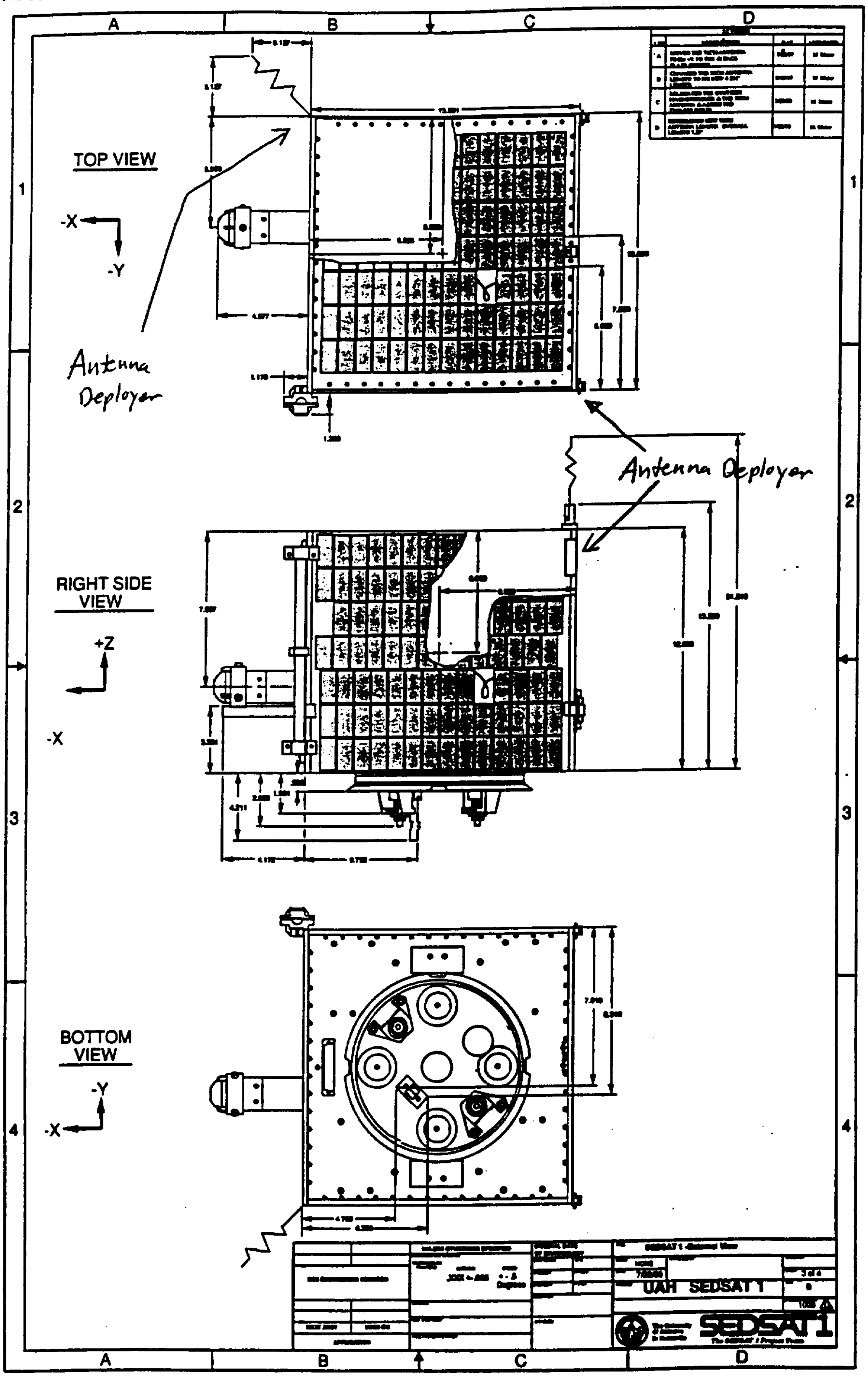


Figure 3.1.2:1 SEDSAT-1 External View (Antennas Not Fully Shown)

3.1.3 SEDSAT-1 Marmon Clamp Interface

Boeing Defense and Space constructed the Marmon clamp interface per original drawings from the McDonnell Douglas Corporation. The materials used conform to the applicable MSFC and GSFC requirements. The Marmon Clamp payload interface was designed as an integral unit as shown in Figures 3.1.3-1 and Figure 3.1.3-2.

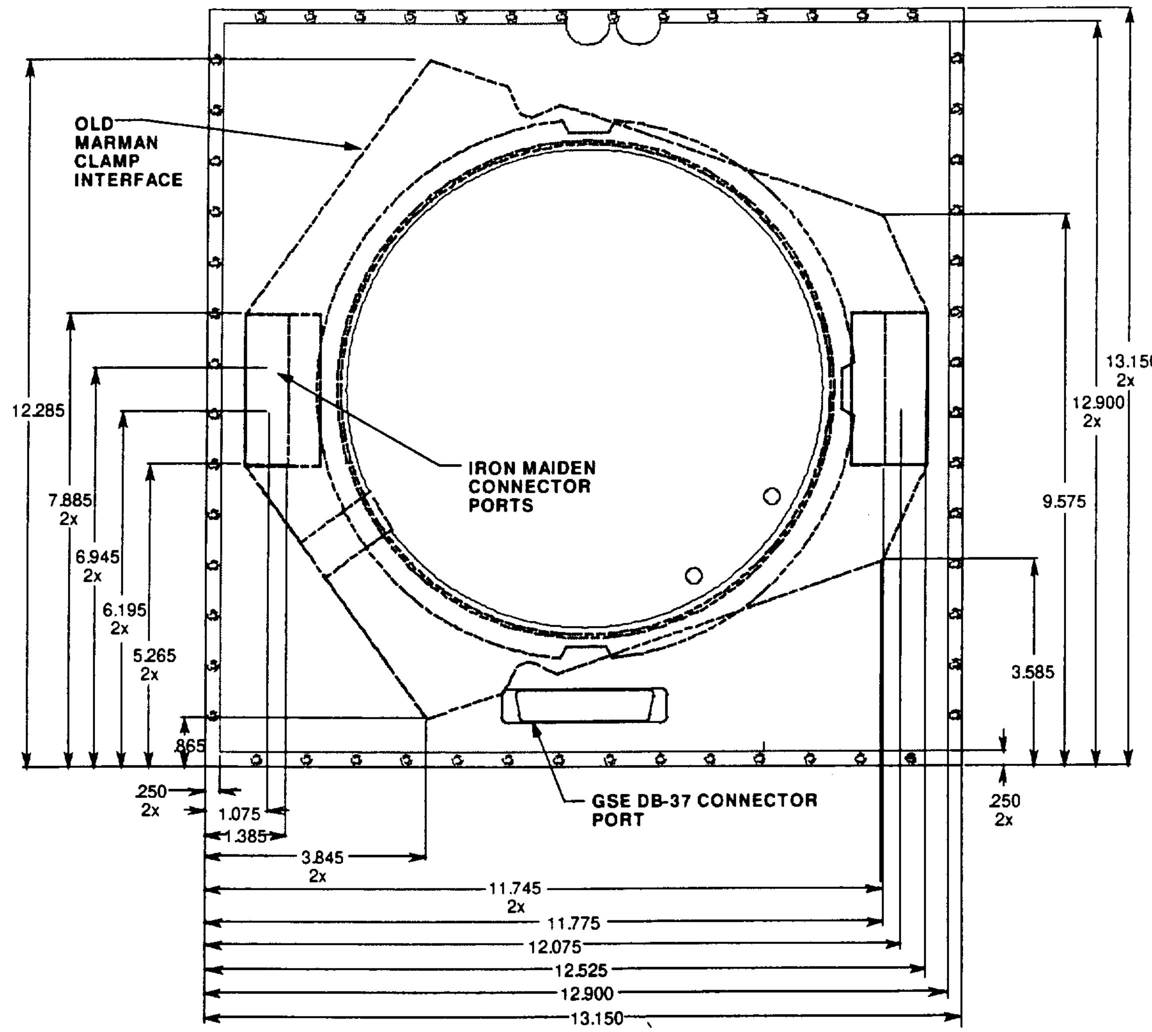


Figure 3.1.3.1 SEDSAT-1 Bottom and Launch Vehicle Interface (top view)

The original Marmon Clamp Interface utilizing a separate payload adapter is outlined within the above plate to illustrate where it would be in relation to the rest of the plate.

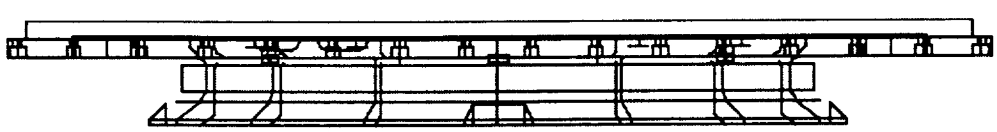


Figure 3.1.3.2 SEDSAT-1 Marmon Clamp Interface (side view)

Not shown in this drawing are the two physical plunger switches that protrude downward and contact the Delta II PA interface. Those switches can be seen clearly in figure 3.1.2-1. The switches are Honeywell commercial 1HE1-6 separation switches procured by NASA Marshall Space Flight Center. These switches were originally designed for Mil-Std-8805. The two switches used on SEDSAT are the two passing members of a set of five subjected to a acceptance test in accordance with the acceptance test requirements of McDonnell Douglas Astronautics drawing 1B99614, Switch, Plunger, 4PDT.

3.1.4 Thermal System

The SEDSAT 1 satellite thermal system is passive. It is radiatively cooled during orbital operations and is passive while attached to the launch vehicle. All internal components are conductively cooled with thermal paths from heat generating elements to the external shell of the satellite. Thermal excursions during orbit are minimal because of the large thermal mass of the satellite and the concentration of that mass in the aluminum structural components.

There are four relatively "hot" spots on the satellite. These are the power amplifiers for each transponder, (two total) and the two antenna deployer thermal actuators. The two transponder power amplifiers have dedicated extra copper heat dissipation paths to the -Y/+Y faces of the satellite. The two thermal actuators are isolated from the satellite to reduce the total heat necessary to actuate during the deployment of the mode-A transponder transmit antenna. The batteries are not considered an important thermal source or sink due to their high efficiency and intimate contact with the -Z face of the satellite, which is the principal radiative face of the satellite.

During launch the satellite temperature is not controlled. The satellite is designed to be safe and operable throughout the temperature envelope of waiting for launch. The temperature excursions during launch were analyzed by Boeing and are documented in A3-L262-LEPT-98-134. The thermal excursions were very small compared to the designed operating range. The maximum temperature estimated was less than 75 deg. F.

3.1.5 SEDSAT Experiment Mounting Plate

The science experiments are mounted in a separate subassembly called the Experiment Mounting Plate (EMP). This plate is structurally independent from the rest of the satellite until final assembly. Figure 3.1.5-1 and 3.1.5-2 show the locations and arrangements of the SEDSAT-1 Experiment Mounting Plate subassembly.

When the EMP is integrated it acts as a structural stiffener. The EMP is 13.15" in the X or long direction. This provides a .25 inch inset into the -X, +X faces of the satellite. The -X and +X faces of the satellite have matching insets.

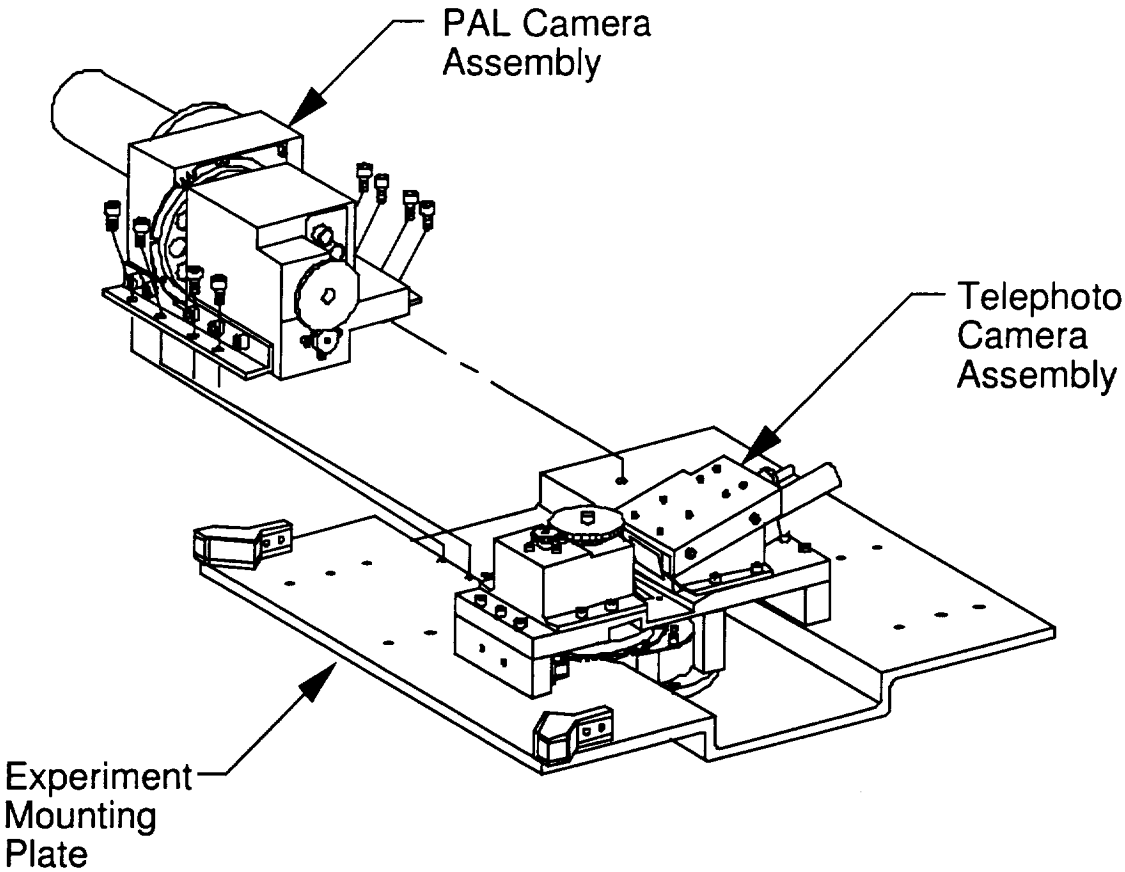


Figure 3.1.5-1 SEDSAT-1 Experiment Mounting Plate Subassembly

Figure 3.1.5-1 SEDSAT-1 Showing insertion of experiment mounting plate

3.1.6 SEDSAT-1 Electronics Subassembly

The SEDSAT-1 electronics subassembly was designed to minimize the number of internal wiring harnesses. The electronics subassembly consists of five printed circuit boards. The top board is the mode A transponder. The second board is the Mode L transponder. The third board is the Command Data System. The fourth board is the SEASIS processor and mass memory board. The fifth is the motherboard. This board acts as the interconnect path between the four other boards. It also acts as the main path for distributing power to the other boards from the batteries and solar arrays. The electronics subassembly is attached to the main structure by wedge locks located on the Y face sides of the boards. Figure 3.1.6-1 is the layout of the subassembly.

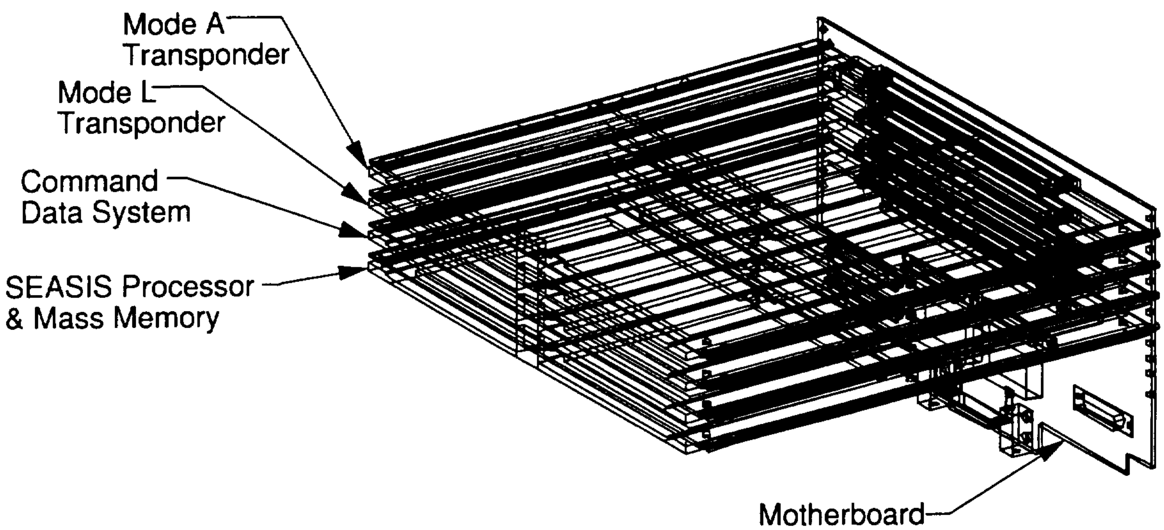


Figure 3.1.6-1 Electronics Subassembly Board Layout

The Electronics subassembly adds to structural stiffness through its installation. The motherboard is fastened across the Y axis elements. The wedge locks provide a thermal path from the multilayer printed circuit boards. The circuit cards ends opposite from the motherboard are inset into the main structure 0.25". All boards are attached to aluminum stiffener frames that also provide additional thermal contacts.

Figure 3.1.6-2 shows the electronics boards being inserted. The SEASIS board is in place while the other slots are empty. 3.1.6-3 shows an intermediate phase of the insertion. Figure 3.1.6-4 shows the final step when the motherboard is pushed against the other board connectors to mate the connectors, and is then secured to the satellite sides.

Figure 3.1.6-2 SEDSAT-1 showing electronics board slots. SEASIS board has been inserted.

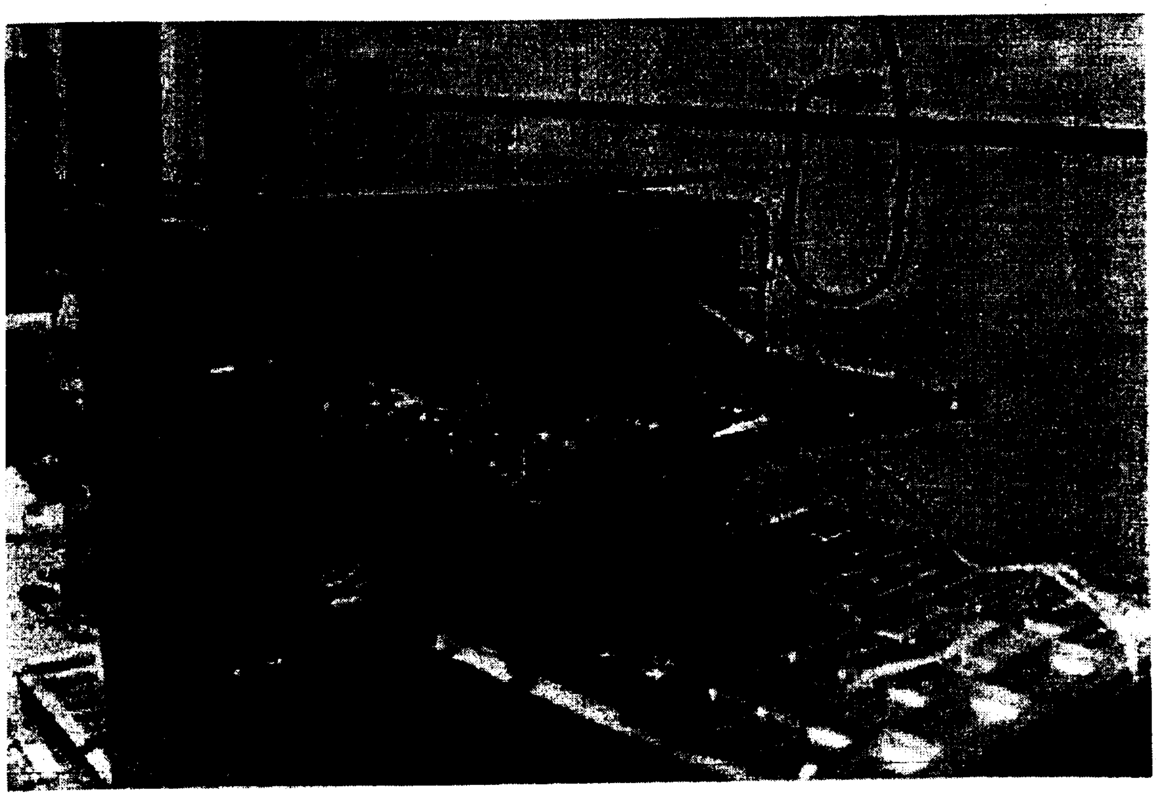


Figure 3.1.6-3 Mode-L being inserted, SEASIS and CDS already in place

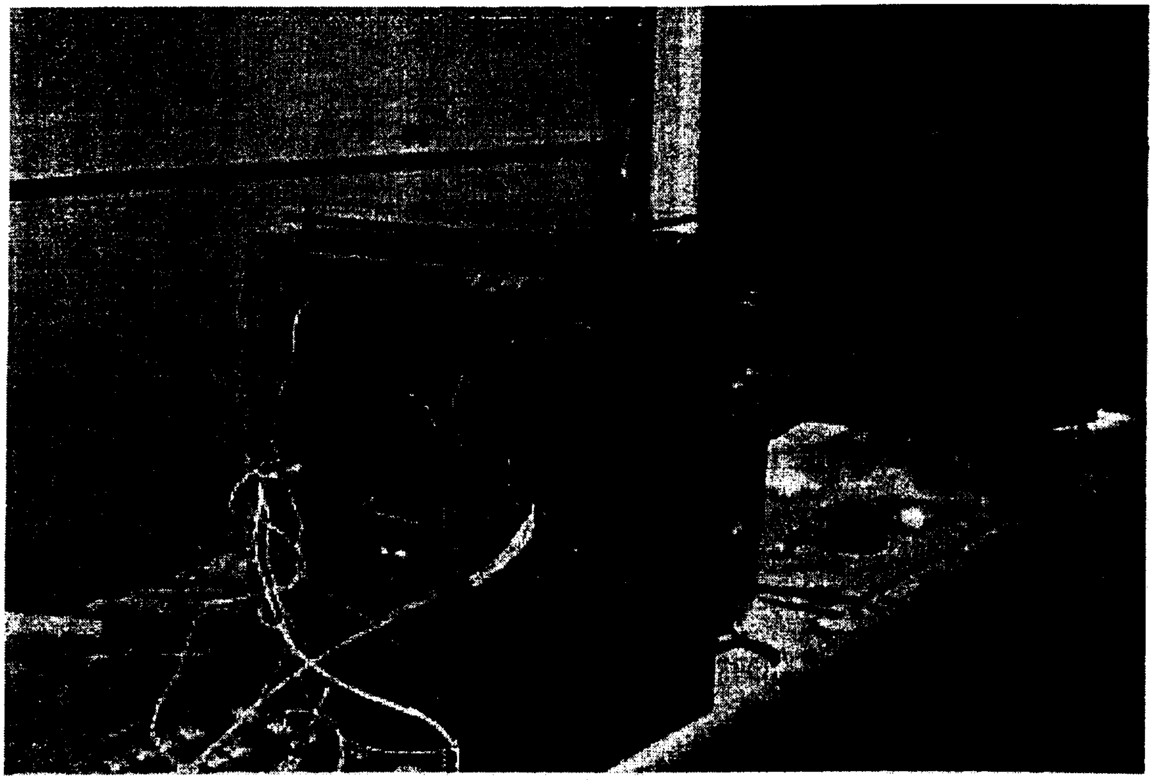


Figure 3.1.6-4 Motherboard placed onto electronics boards in last electronics assembly step

3.1.6.1 SEDSAT-1 Antennas

There are two sets of antennas (Mode-A and Mode-L) on SEDSAT-1 comprising a Mode-L transmit, Mode-L receive, Mode-A receive, and Mode-A transmit pair to support amateur radio satellite communications. The 70 cm wavelength (6 inches long) transmit monopole, the 2 m wavelength (19 inches long) receive monopole, the 23 cm wavelength (2.56 inches long) receive monopole, and the 10 m wavelength (8 feet long) dipole pair are all omni-linear antennas. Figure 3.1.6.1-1 shows the antenna configuration.

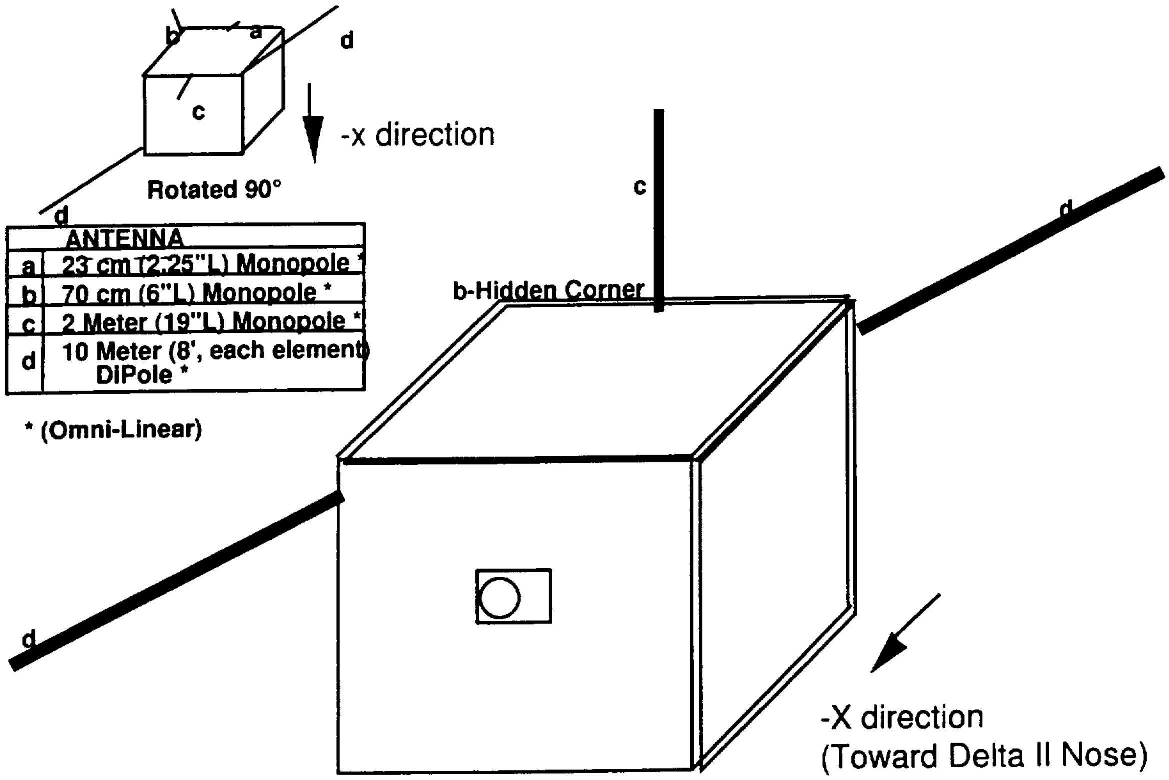


Figure 3.1.6.1-1 SEDSAT Antenna Configuration (After full deployment)

The antennas are made of spring steel from Stanley tape measures. This material is very common for use in Microsatellites. The table below gives details of the antennas.

Antenna Length (INCHES) and Weight (Pounds)

	Length (In.)	Weight (lb.)
a. 23 cm Monopole, Fixed	2.25	.1
b. 70 cm Monopole, Fixed	6	.05
c. 2 Meter Antenna, Monopole, Fixed	19	.1
d. 10 Meter Antenna, Deployed	96" each	.5

The fixed antennas are secured to the satellite by machined pieces of Delrin black with 440 metal fasteners.

3.1.6.2 SEDSAT-10 Meter Antenna and Deployer

The only deployable antennas on SEDSAT-1 are the two 10 meter band antennas. The antennas are each 96" in length and are made from the same Stanley tape measure material as the fixed antennas. After the satellite is deployed from the Delta II and is in its final orbit, the antennas will be deployed. The antennas are stored furled in the antenna deployers, held in by a swinging door that is prevented from moving by a latch. The latch is rotated open by thermally activated actuators. After the door opens each antenna will deploy from the force of its own spring tension.

The actuators are a thermal paraffin type that has a stroke of approximately 0.4 inches. The paraffin is heated to its melting point by a Kapton Insulated Flexible Heater. The melting paraffin expands, pushing the actuator piston its stroke distance. The process of heating the paraffin to a temperature hot enough to cause the actuator to travel its stroke length takes approximately 10 minutes at a heater current of 1 amp. The part is made by Actronics Incorporated and the part number is P/N 11-546. This actuator has been designed into other NASA missions and the certification of compliance with NASA and military standards are on file at NASA MSFC and UAH. The specified actuation temperature is no movement below 180 deg F and full stroke at 195 deg F.

The antenna deployment mechanisms are located within the top two inches of the satellite. They are mounted on a .125 " thick plate that is located above the electronics subassembly. Figure 3.1.6.2-1 shows the placement of the antenna canisters on the mounting plate.

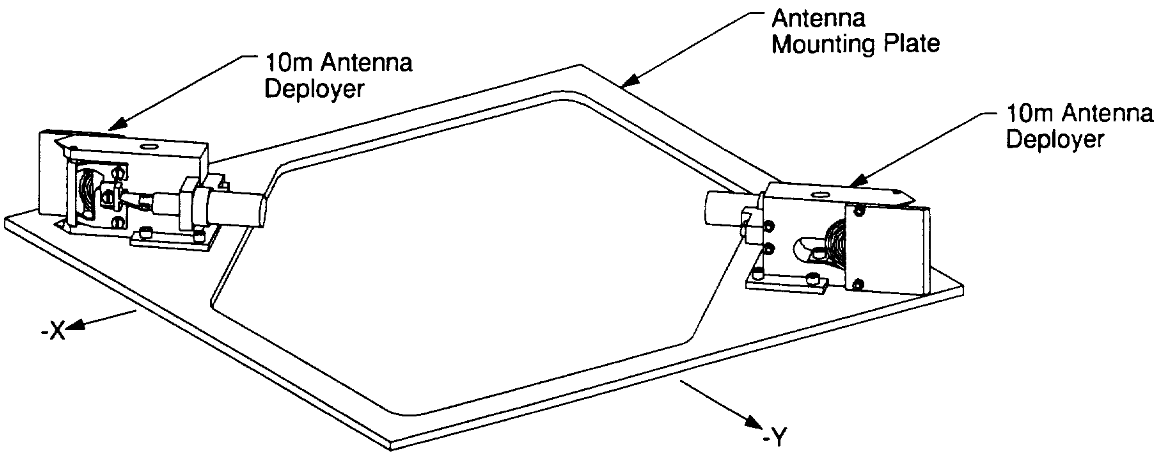


Figure 3.1.6.2-1. Antenna Mounting Plate With Antenna Deployers

The weight of the antennas is 0.067 lbs. each. The closing force of the deployer doors is 1.52 Newtons. The antenna deployers receive power from the main power bus through an Interpoint DC/DC converter and a bank of power MOSFETs, which are controlled by a CDS controlled multiplexer. The deployers are powered on by the CDS board activating the DC/DC enable through software and selecting the deployer power MOSFET through the mux. In order to activate the deployers the satellite must power on (which is inhibited as described below) and the CDS provided converter enable and mux signals must go high. The schematic is shown in figure 3.1.6.2-1. The deployers, like other onboard electronics, are not monitored on the ground because they are unpowered.

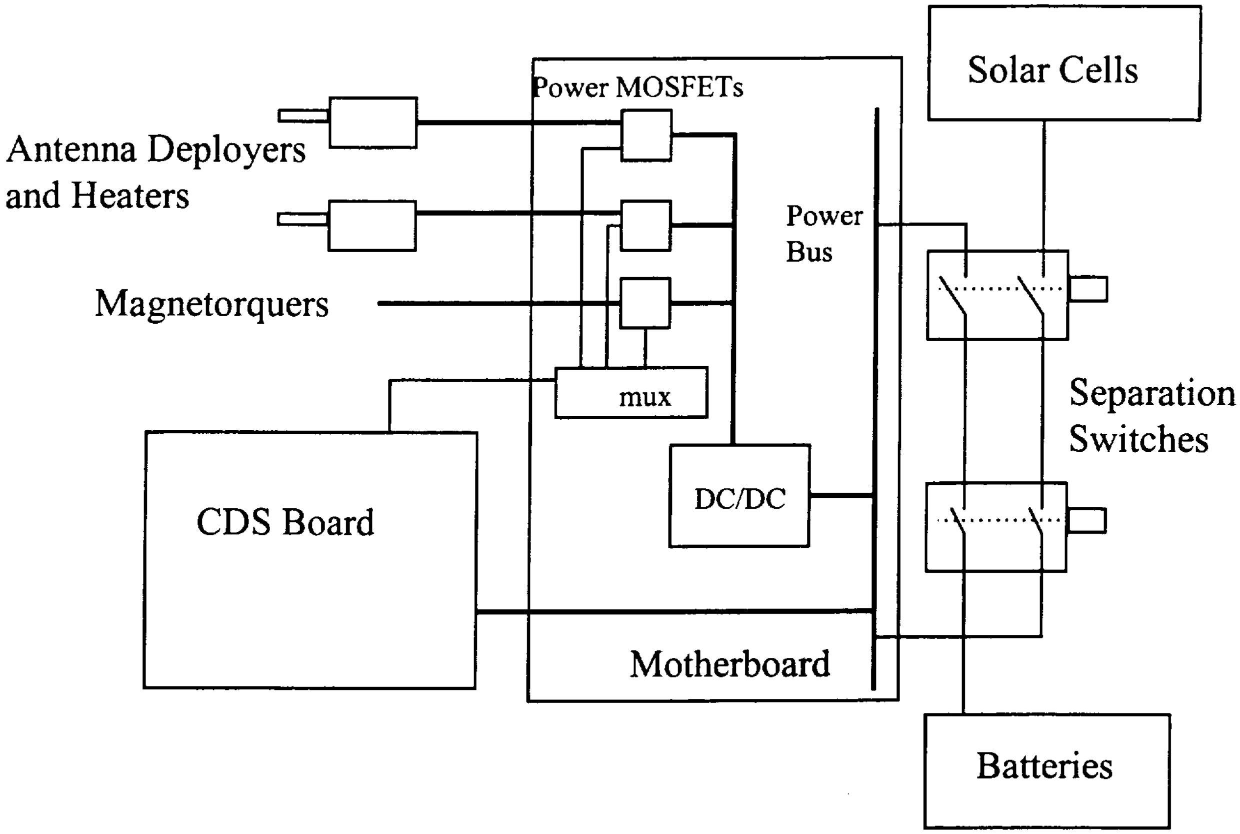


Figure 3.1.6.2-2. Antenna deployer schematic

3.2 SEDSAT-1 Electrical Power System

SEDSAT-1 does not have a centralized power regulation system. Each subsystem has its own DC/DC converter to power its functions. The layout of the power system is shown in figure 3.2-1. This figure also shows the terminals available at the GSE ports, allowing verification of the status of the separation switches, external battery charging, and enabling and disabling of the satellite. There are a total of six Interpoint DC/DC converters used in SEDSAT-1. With a wide input range (16 to 40 volts) these converters regulate to within 1% of stated output voltage. Several advantages to this design are:

1. Simplified power supply design. (No central custom supply to build)
2. Simplified wiring harness. (Only main power distributed)
3. High Efficiency DC/DC converters space qualified and available inexpensively.
4. DC/DC converters incorporate power inhibits to simplify power control.
5. Some fault tolerance to subsystem overcurrent failures since individual DC/DC converters can isolate the remaining subsystems.

The schematic of the power distribution system is shown in figure 3.2-1 and is further described herein. Inadvertent power on of hazardous subsystems is prevented by two means. First, power is redundantly inhibited through the separation switches once the PAF is attached. Second, power is inhibited by safety plugs (or by the lack of any plug) in the GSE ports during all handling during which the separation switches are allowed to close. While safety plugs can be used, the configuration of the GSE ports is such that the system is inhibited from powering on unless flight jumpers are inserted. The basic safety strategy is to keep the system unpowered until flight separation. In addition, the system can only be commanded to perform some hazardous operations (such as antenna deployment) by a relatively complex communications protocol.

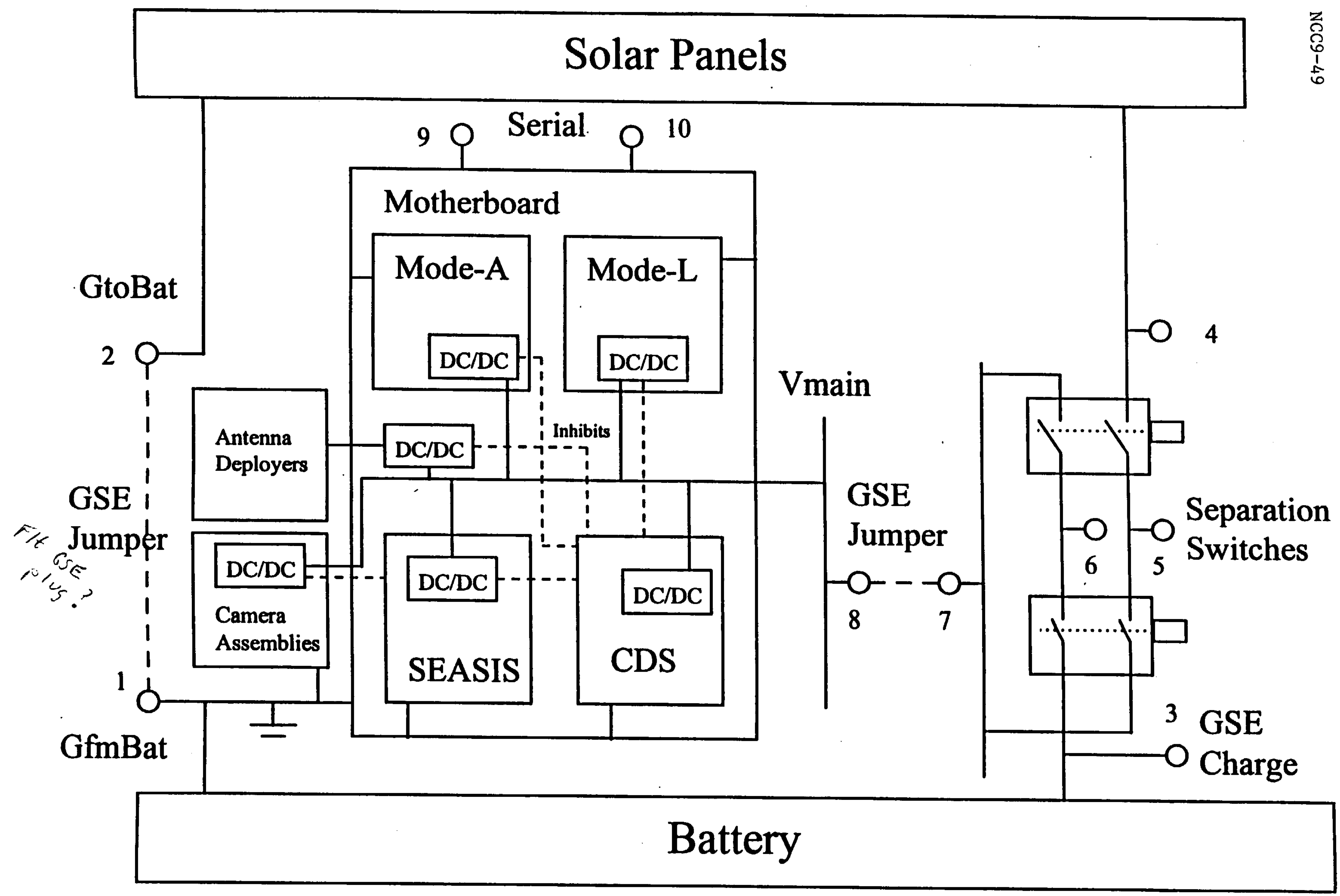


Figure 3.2-1: Power system schematic

While in an unpowered state with the batteries fully charged there is a potential shock or sparking hazard at points that are battery energized. These hazards are ameliorated through a variety of design practices. Since GSE port pins are powered, the GSE port connectors are female only. Internally, powered sections are in close proximity only at the separation switches, which are hermetically sealed.

The RF radiation hazard is inhibited only by keeping the spacecraft in an unpowered state. If the separation switches are allowed to close while a flight jumper is in place the system will begin to radiate, and will not give any external indication that it has done so. This issue is controlled by monitoring the state of the separation switches via electronic measurement during and after PAF installation and at any other time until the flight jumper is installed. The flight jumper is installed on only one GSE port. It is possible to monitor the spacecraft from the other GSE port at any time, but only by attaching a cable and GSE (the switch/fuse box) to the satellite.

The numbered terminals in figure 3.2-1 redundantly map to the GSE port pins. The pin mappings for the GSE ports and the plugs used are shown in Table 3.2-1.

Port A		Port B		Flight Jumper	
Pin Number	Fig 3.2-1 Terminal	Pin Number	Figure 3.2-1 Terminal	Port A	Port B
1	1	1		2 to 3	2 to 3
2	9	2	7		6 to 7
3	10	3	8		4 to 5
4	4	4	2		8 to 9
5	5	5	1		
6	6	6	7		
7	7	7	8		
8	3	8	2		
9	8	9	1		

Table 3.2-1: GSE Port pin mapping

The charging plug brings all Port-A pins to the switch/fuse box. Current is applied to pins 1 and 9 (corresponding to terminals 1 and 8).

3.2.1.1 Electrical Power System and the Interface to the Delta II

There is no electrical interface between SEDSAT-1 and the Delta II rocket. SEDSAT-1 electronics are unpowered until deployment. The batteries are charged and the solar cells may be active, but power from both is inhibited from the electronics by the separation switches. The separation switches are single pole, dual throw plunger switches (Honeywell part 1HE1-6). Each separation switch incorporates two isolated, single pole, dual throw switches actuated by a common plunger. The plunger seats on a spring that provides enough force to extend the plunger. When the plunger is extended one of the switch poles is engaged. When the plunger is compressed the switch pole is disengaged, and the opposite pole is engaged when plunger travel is nearly complete. SEDSAT uses two of the Honeywell switches connected in series to provided single fault tolerant inhibiting of the SEDSAT electronics. The separation switches are mounted inside the Marmon clamp ring with the plunger protruding below the interface ring (shown in figure 3.2.1.1-1). When SEDSAT is mated to the payload attach fitting (PAF) the separation switch plungers are pressed against the surface of the PAF. The switches are mounted so that the plunger depression against the PAF is nearly equal to full travel. Figure 3.2.1.1-2 shows the configuration with the PAF attached.

Only the plunger extended switch pole is used on SEDSAT. The depressed pole is often used in other applications of the Honeywell switches, but is not used on SEDSAT. The plunger extended poles are wired into the SEDSAT electronics.

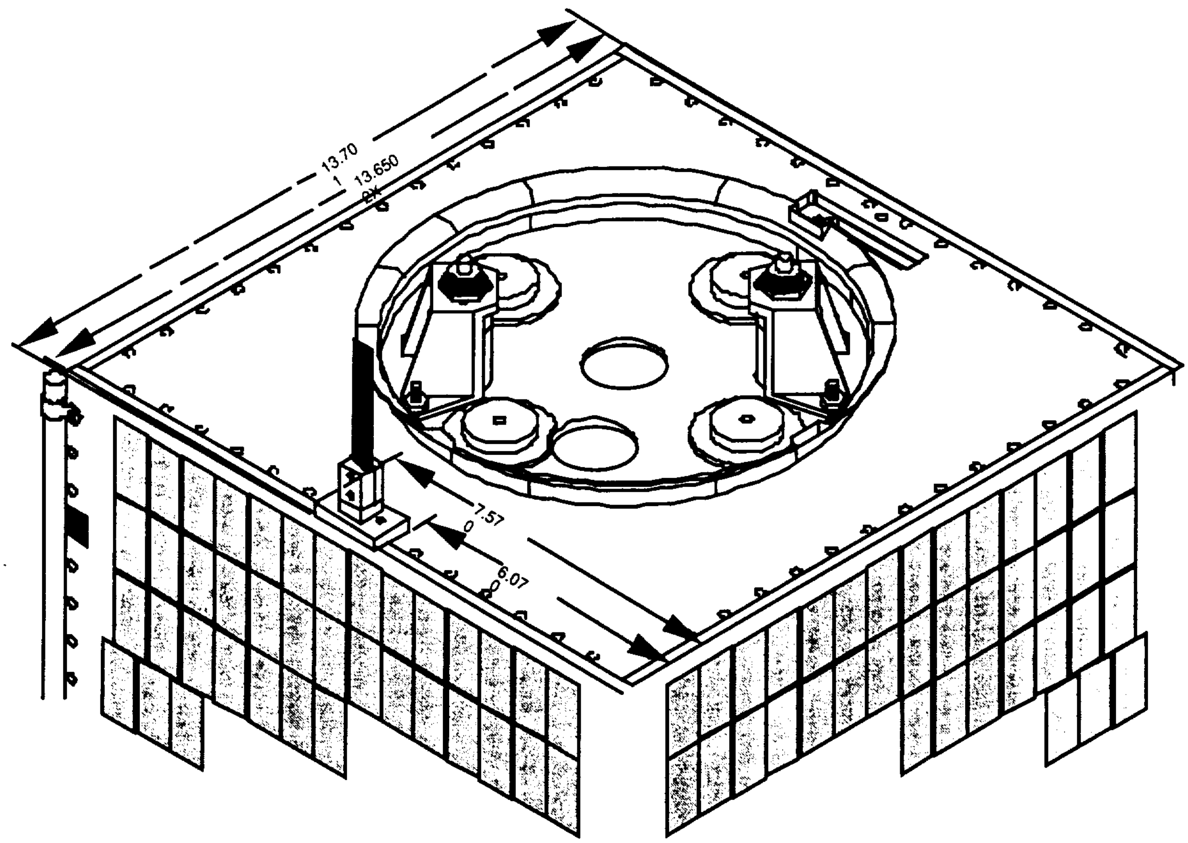


Figure 3.2.1.1-1 Seperation switches on –Z face plate

Electrically, the switches disengage both the batteries and the solar arrays from the satellite. Battery positive and solar panel positive both pass through the two switches (in series) before reaching each other on the main power bus. Both separation switches must fail closed to allow power from either the batteries or solar panels to reach the electronics, or each other. The negative sides are separated from each other until the flight GSE plug is installed, and the main power bus is separated from the electronics by the same plug. This arrangement is shown in figure 3.2-1.

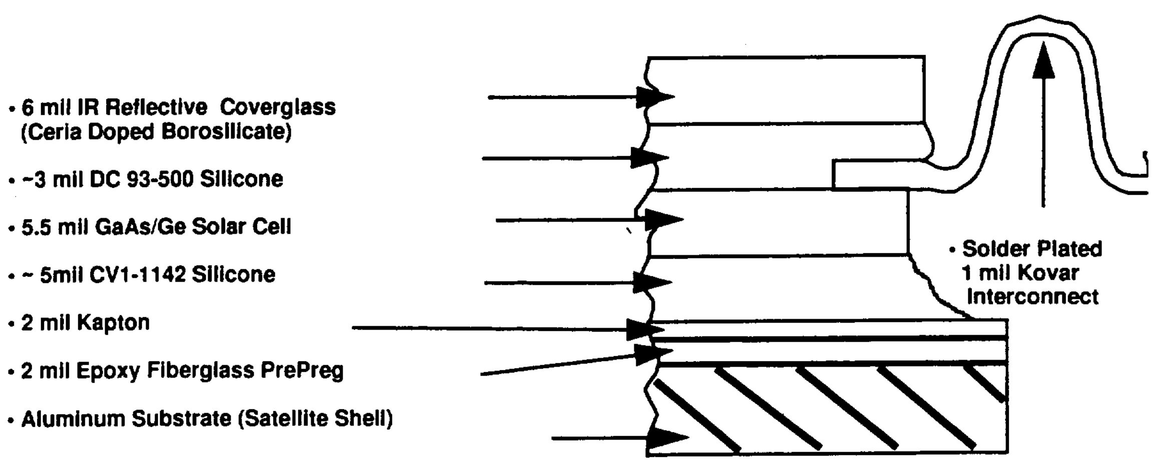
At the moment of ejection the plungers are released, and the switches move to the wired poles connecting power to the satellite. The arrangement of the power system with the physical switches is dictated by the Secondary Payload Planner's Guide for requirements for single fault tolerance to insure non-operation before separation. Figure 3.2-1 is the schematic for the separation switch inhibit mechanism. Also included is the GSE interface that allows the bypass of the switches for ground testing, verification of safety interlock operation, and charging of the batteries after closeout of the satellite structure.

The Honeywill 1HE1-6 separation switches were selected in a compromise between availability and qualification. Honeywell has previously supplied a separation switch built to a McDonell-Douglas specification and built to a reliablity program. Unfortunately, these switches are no longer manufactured and no alternate switch with a full qualification and reliability program will be offered in the foreseeable future. Instead the decision was made to use a hermetically sealed, Mil-8805 qualified switch put through the same acceptance test as called for in the previous McDonnell-Douglas specification. Of five 1HE1-6 switches put through this test program two passed. The three failing switches failed hermetic seal but did not fail in operation. The two passing switches are used on SEDSAT-1.

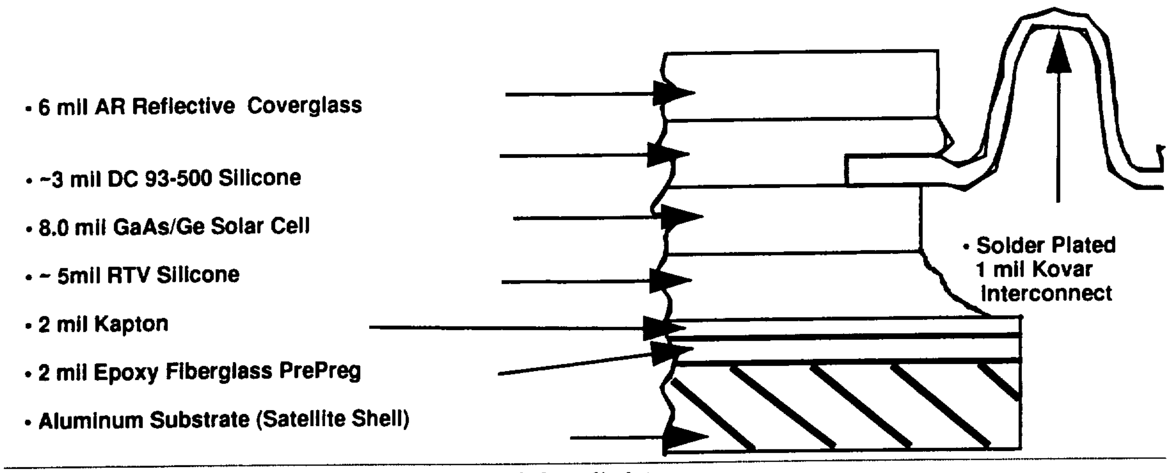
Figure 3.2.1.1-2 SEDSAT-1 attached to PAF with Marmon clamp installed

3.2.1.2 Solar Arrays

The solar arrays for SEDSAT-1 are GaAs/Ge arrays and were provided by TRW and Applied Solar Energy Corporation. On each of the five faces there are three strings with 32 cells in series on the TRW arrays and 35 cells per string on the Applied Solar Energy arrays. Each string is diode protected from shadow induced reverse currents by low loss diodes located on the motherboard. The wire gauge for each string is 26 and complies with appropriate Delta standards. The arrays were assembled on the SEDSAT-1 structure to reduce manufacturing costs and pointing requirements for the arrays. Figure 3.2.1.4-1 shows the solar arrays and how they are assembled on the satellite structure.



TRW Supplied Arrays



ASEC Supplied Arrays

Figure 3.2.1.4-1 SEDSAT-1 Solar Array Materials and Construction

The coverglass for the solar arrays comprise two different types. The TRW supplied solar arrays, (+Z, +/- Y faces) use an IRR coverglass material that is reflective in the infrared and will work to lower the temperature of the solar arrays and the satellite. The Applied Solar Energy supplied solar arrays use the standard CMX coverglass.

3.2.1.3 Battery

The SEDSAT-1 battery is the source of energy to operate the satellite during the shadow portion of the orbit. The batteries use 8 Ah Nickel Metal Hydride (Ni-MH) cells. There are 16 cells total with eight cells packed into each of two battery boxes. The cells are Nickel Metal Hydride chemistry containing 32 cc per cell of 1.3 specific gravity potassium hydroxide electrolyte. The battery boxes have two vent holes that also provide for routing of the wiring into and out of the box. A venting analysis including the effects of the wiring has been performed and shows less than .001 psi pressure differential during Delta ascent. Figure 3.2.1.5-1 illustrates the battery box design for containing the Ni-MH cells.

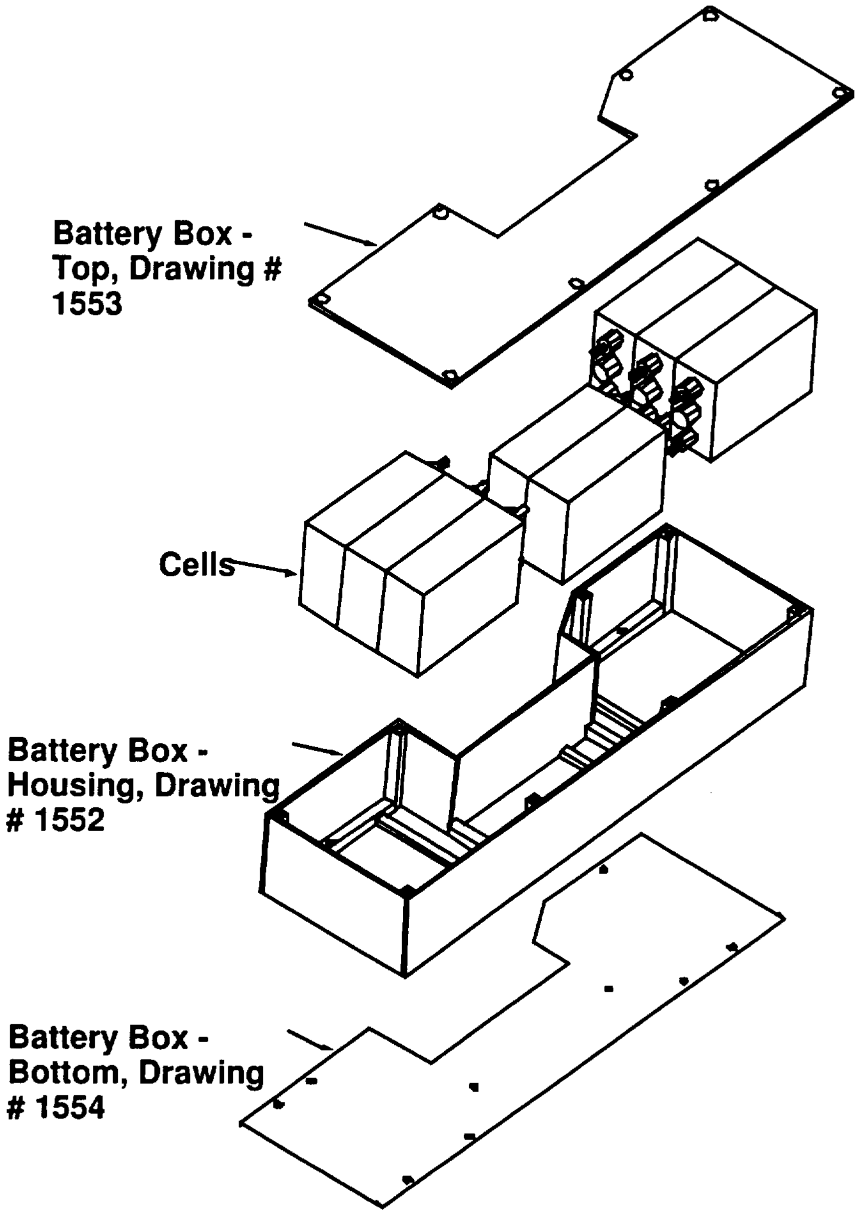


Figure 3.2.1.5-1 SEDSAT-1 Battery Box (2 each)

The cells themselves are prismatic cells with the same specifications as military standard NiCad cell prismatic packaging. These cells have a relief valve installed to vent in the event of severe overpressure. The valves operate at 180 psi +/- 20 psi. The burst pressure for these batteries is approximately 600 psi. Test information regarding failure of the battery has been obtained and can be provided through the MSFC battery laboratory. The battery boxes are seen during assembly in Figure 3.2.1.5-2.

Figure 3.2.1.5-2 Battery boxes seen below experiment mounting plate during assembly

These batteries do involve a safety hazard and required extensive review. One consideration for safety is trickle charging. Trickle charging was planned only for the SSPF and so did not involve hazards to the Delta II or primary payload. At the specified trickle charge level of 23 volts and current limited to 0.1 amps the batteries do not rise in temperature enough to pose a hazard. As an example, a duplicate set of flight batteries were trickle charged in an insulated container to and beyond full charge. The temperature history is shown in figure 3.2.1.5-3. Charger overcurrent is inhibited by manual setting of the charger power supply, by charger box fuses, and by monitoring.

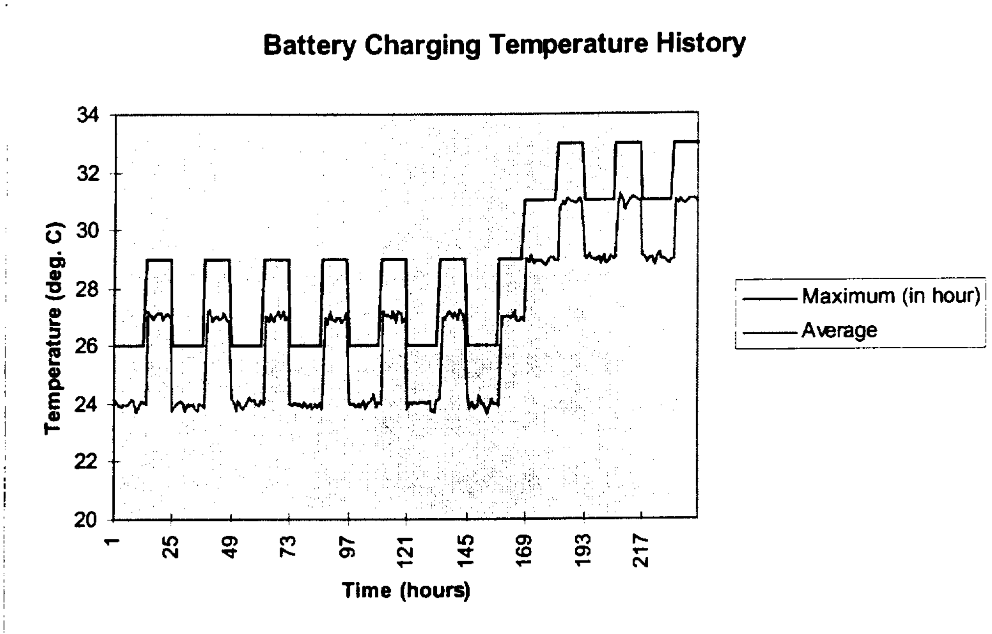


Figure 3.2.1.5-3 SEDSAT-1 Trickle charge battery temperature history

3.3 SEDSAT-1 Communications System

The communications system for SEDSAT-1 is intended for the use of amateur radio satellite operators worldwide and transmission of scientific data. There are two transponders on board. The primary transponder is a Mode L transponder as defined by the International Amateur Radio Union. The second is a Mode A transponder that will be used as a backup to the primary transponder. Table 3.3.1 gives the characteristics of the communications system and a summary link budget analysis.

Parameter	Mode-A (Analog repeater)	Mode-L (Digital)
Frequency		
Uplink	145.915 to 145.975	1268.175 to 1268.250
Downlink	29.35 to 29.41	437.850 to 438.000
Maximum power (EIRP)	10 W	5 W
Average power, W	8 W	3 W
Type of Transmitter	Class AB Linear	Class AB Digital
Antenna Gain	0	0
Antenna Location	+Z and deployed	-Z and XY side
When operated	On orbit after separation and software command (No RF on pad or SSPF)	Tests in SSPF, on orbit after separation (automatic), no RF on pad
Link Budget		
Nominal Range, km	1000	1000
Downlink rec. ant. Gain, dB	0	5
Downlink Received Power, dBm	-82	-103
Uplink Transmit power, W	10	10
Uplink Trans. Ant. Gain, dB	5	10
Uplink Received Power, dBm	-91	-104

Table 3.3.1 SEDSAT-1 Communications System Characteristics

Inadvertent radiation is inhibited by removing the flight plugs from the GSE ports, and by the compression of the separation switches during PAF installation. Communications system tests were conducted during development, after closure of the case, and before delivery for launch. On orbit the downlink has worked exactly as predicted. However, no uplink attempts have been successful.

Safety hazards associated with the SEDSAT-1 communications system are minimal. During ground operations the primary hazard is personnel exposure during planned radiation. Because of the low power of the system this hazard is easily controlled by maintaining adequate separation distances (specified distance is 1 meter).

3.3.1 Mode A Transponder

The mode A transponder is derived from the successful mode A transponders flown by AMSAT in the mid 1970's. The frequencies of operation and the system configuration is shown in Figure 3.3.1-1

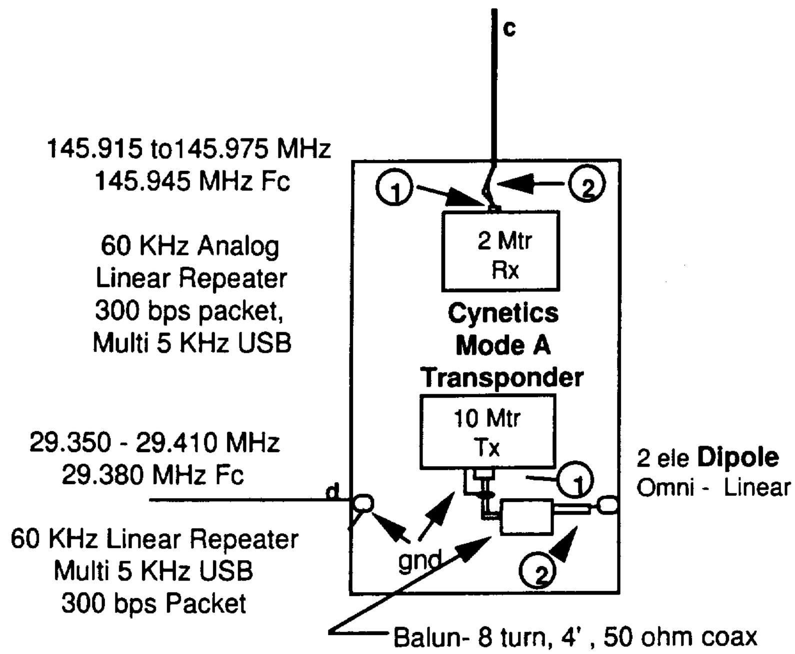


Figure 3.3.1-1 Mode A Transponder System Configuration

The uplink is in the amateur radio 2 meter band. The downlink is in the amateur radio 10 meter band. The primary use of this transponder is for voice communications between amateur radio operators within the RF footprint covered by the satellite. A secondary 300 bit per second data link that is compatible with terrestrial HF packet radio communications is included as a back up to the primary mode L transponder in the event of its total failure. This transponder was not powered on during ground operations or while attached to the Delta II. The Mode-A transmit antennas were not deployed until SEDSAT. The Mode-A transponder cannot be powered until after the antennas have been deployed. Deployment requires successful uplink, and so has not been performed successfully.

3.3.2 Mode L Transponder

The mode L transponder on SEDSAT-1 acts as the primary communications system for the satellite. The transponder communicates at a maximum data rate of 56 kilobits/sec. This will be the primary method of obtaining telemetry and experiment data from SEDSAT-1. Figure 3.3.2.1 outlines the frequencies of operation and configuration of the system.

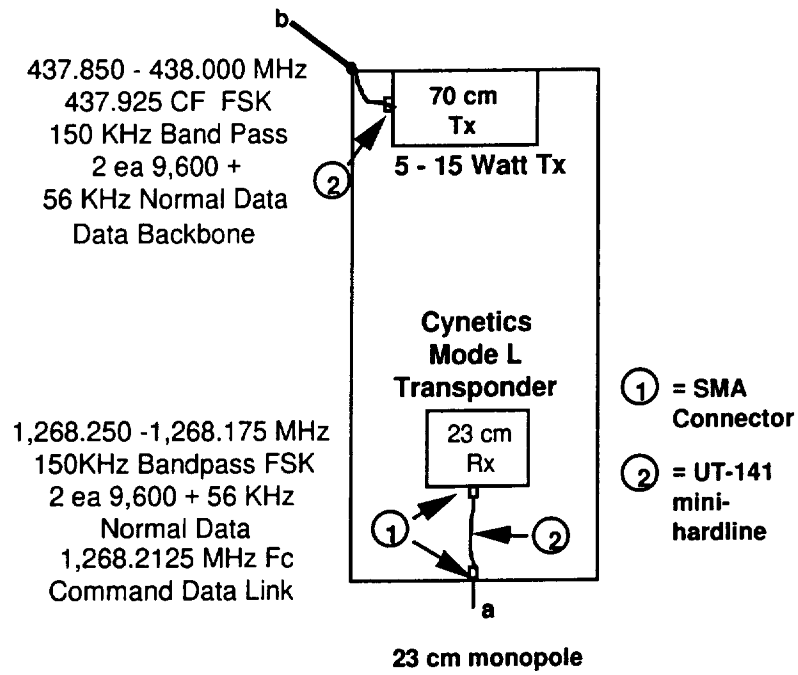


Figure 3.3.2-1 SEDSAT-1 Mode L Transponder System Configuration

This transponder was not active from delivery until SEDSAT was ejected from the Delta II. The transponder was active during ground operations Operation of the transponder during test is described in UAH-SEDSAT-1803.

3.4 SEDSAT-1 Command Data System

The Command Data System (CDS) comprises the bulk of SEDSAT-1's control electronics. The command data system (CDS) design closely parallels the AMSAT UK UoSat design in order to retain as much software compatibility as possible. This design consists of an Intel 80C186 microprocessor, two Intel 82C530 Serial Communications Controllers (SCC's), 2 megabytes of 12 bit error detecting and correcting memory and 48 analog inputs to the A/D converters.

The objective of the Command Data System (CDS) is to serve as the command processor for SEDSAT-1. The functions that the CDS controls are as follows:

1. Command uploading of new programs from the ground.
2. Serial communications through the digital and analog transponders.
3. Electrical power system control.
4. Record telemetry from the TAS and SEASIS experiment.
5. Watchdog Timer for system reset on loss of software control.

3.5 Attitude Control System

Attitude Control of SEDSAT-1 is accomplished actively by the use of electromagnets controlled by the Command Data System. There are two sets of electromagnets. One set is aligned along the Z axis of the satellite and the other set is aligned along the X/Y axis. This system is not enabled until SEDSAT has completely ejected from the Delta II rocket and an uplink established. The eventual objective is to stabilize the satellite with the –Z face pointed toward the earth.

Figure 3.5-1 shows the installed Z direction magnetorquer rod. This rod was the source of some concern because it was found to be broken after the initial vibration and shock tests. It is not known when the break occurred, and whether it was during testing or transportation. A repaired rod was re-installed at Cape Canaveral. As a precaution the rod was encased in a thick shrink-wrap tube before installation. The hole in the center support was also widened to avoid any center torque.

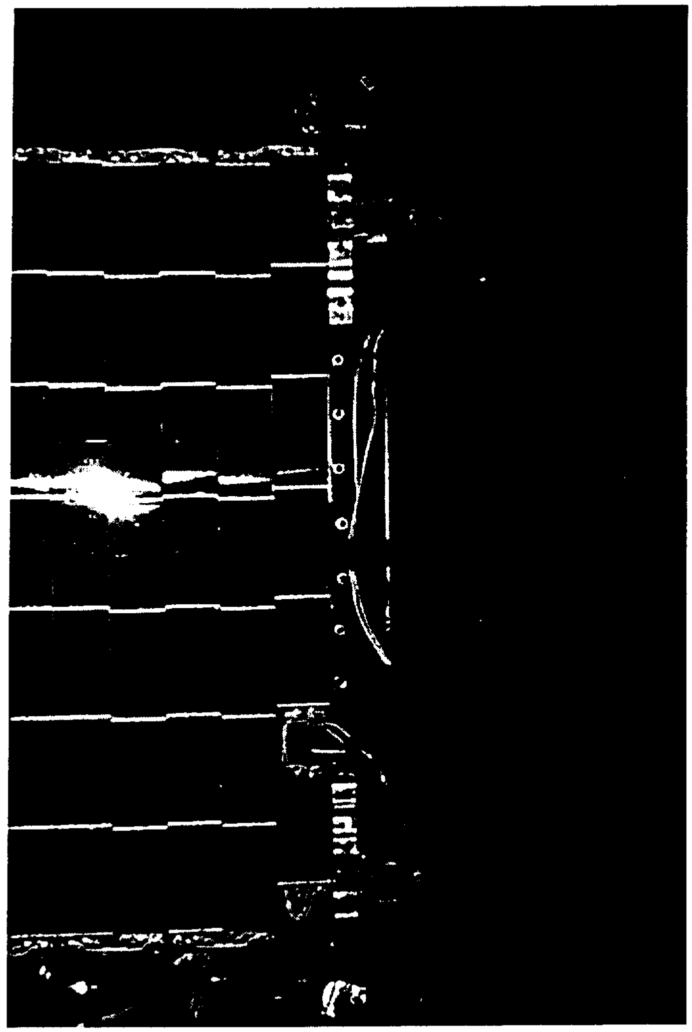


Figure 3.5-1 Z axis magnetorquer rod installed

3.6 *SEDSAT Experiments*

3.6.1 SEASIS

SEASIS has two imaging subsystems that share processing and control electronics (Figure 3.6.1-1). The telephoto system has a telephoto lens with a 10 degree field of view that will be pointed nadir (or in the nadir direction) after SEDSAT has reached its final orbit. The Panoramic Annular Lens (PAL) imaging system will be pointed 90 degrees away from zenith or nadir. Both systems have a filter wheel which has 6 narrow band filters, 4 broad band filters, and 2 neutral density filters. They also have a Sony CCD camera that will capture the images focused onto the image plane by the lenses. SEASIS has redundant video digitizers that will digitize images from the cameras mounted on the Transputer based SEASIS processor/memory subsystem. The Transputer is the computer chip used for the SEASIS electronics.

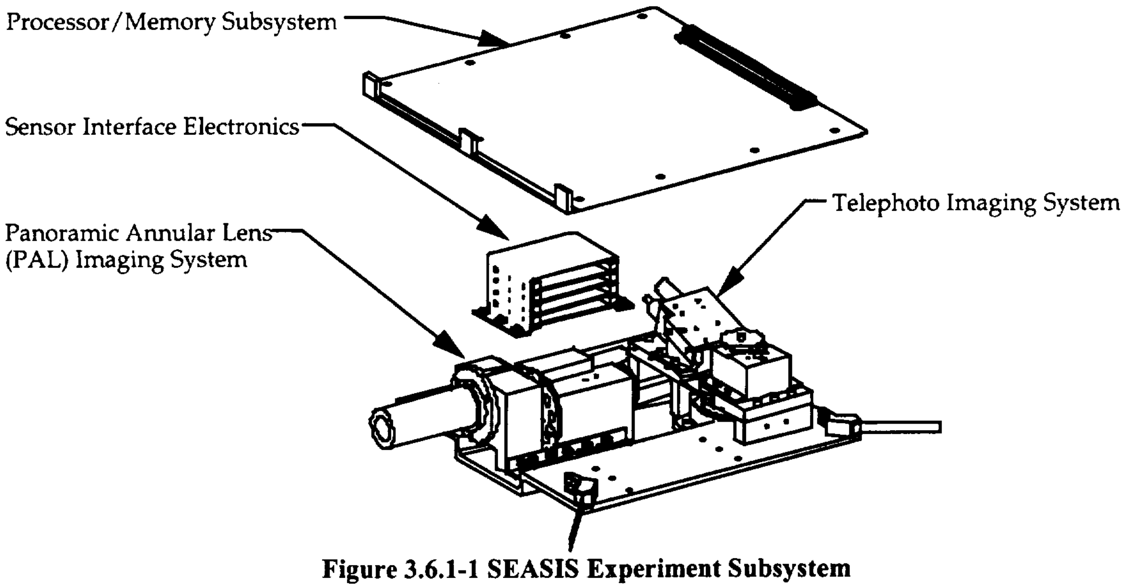


Figure 3.6.1-1 SEASIS Experiment Subsystem

Figure 3.6.1.2-1 shows a functional block diagram of SEASIS. For both the telephoto system and the PAL system, light is focused by a lens assembly, through a bandpass filter, onto the image plane of a CCD camera. The video digitizer records one image at a time, from either the telephoto camera or PAL camera, digitizes it, and performs a direct memory access (DMA) transfer to the Transputer. The Transputer compresses/processes the digitized image and transfers it to the SEASIS memory subsystem, currently designed for 1 Gigabit of static memory. Images are stored in the memory subsystem until requested by the Command Data System (CDS), the satellite master control system, for down link to earth. The images are transferred by the Transputer from memory, through the processor module, to the CDS . The CDS formats the images for transmission, and then down links the images via a Mode L transponder at 437.85 MHz with a data rate of 9.6, 19.2, 38.5, or 56.0 kbits per second. Two Interpoint DC/DC converters provides power to the experiment from the main satellite power bus. One is for the electronics and the second is for the cameras and stepper motors. The physical camera assembly is shown in figure 3.6.1.2-2.

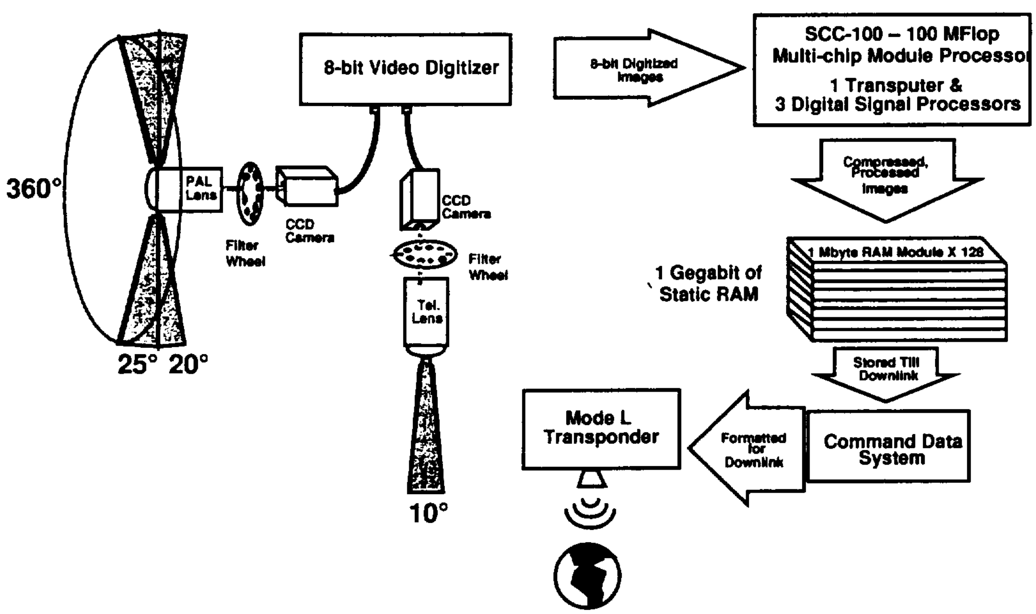


Figure 3.6.1.2-1 SEASIS Functional Block Diagram

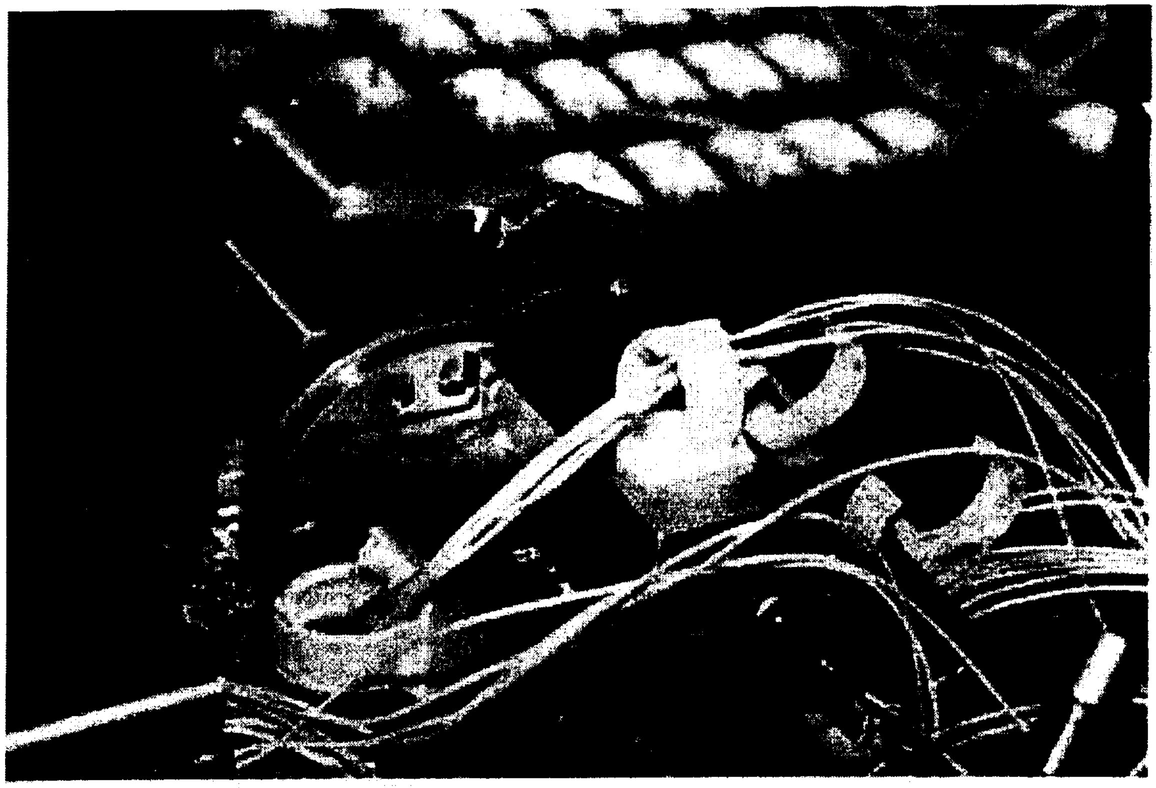


Figure 3.6.1.2-2 SEASIS camera assembly (PAL camera)

3.7 Mission Operations

SEDSAT 1 mission operations commenc when the satellite is ejected from the Delta II rocket, separating from the PAF. Separation from the PAF frees the separation switch plungers allowing internal springs to move them to their unloaded position. This closes switches internal to the separation switches, which connect the solar arrays and the battery to the main power bus. There is a 1 second time constant RC circuit that prevents the CDS 80C186 processor from operating during power transients that arise from the abrupt applying of power during switch closure.

After the processor comes out of its reset cycle, it runs through an initialization cycle where the memory is configured by the real time multitasker (kernel). This initialization cycle lasts for approximately 1 second whereupon the kernel begins normal operations by beginning the telemetry capture task which includes recording and processing all telemetry from the power system. The SEASIS system is activated to take images of the receding Delta II second stage. After this task is successfully started, the communications system is initialized and the mode L transponder commanded into its high data rate mode with telemetry, including image data transmitted.

Once ground communications is established new flight software will be uploaded. The onboard software image is limited by constraints of the operating system, but can be readily replaced by software transmitted from the ground. However, as noted before, attempts at uploading so far have failed.

During the mission, the SEDSAT-1 ground team in association with amateur radio organizations will have operational ground stations at many points around the world. A subset of these stations will have Internet connectivity, which will pipeline the data from their radios. This information will be fed to the University of Arizona for retransmission to POCC at UAH and to any other Internet site with the appropriate software. This plan has been largely accomplished in conjunction with the National SEDS chapter at the University of Arizona.

3.8 *Integration and Test*

The SEDSAT qualification and acceptance plan was driven by the Boeing secondary payload requirements as well as engineering needs. Developmental testing was done in the NASA Marshall Space Flight Center thermal-vacuum chamber. Acceptance testing, using proto-flight methodologies and levels, was also performed at NASA Marshall.

3.8.1 Thermal-Vacuum Test

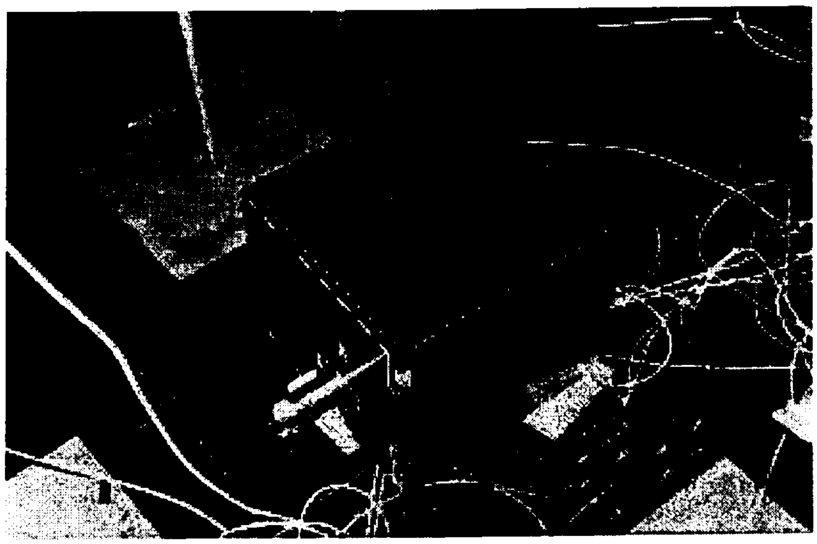


Figure 3.8.1-1 SEASIS Functional Block Diagram

Thermal-vacuum tests were run in August, 1997. These were developmental tests that concentrated on showing stable system operation. The tests included deploying both halves of the antenna under cold conditions and evaluating thermal balance. As a result of the tests the installation of the antenna deployers had to be modified to reduce heat conduction. All other parts of the satellite were found to be satisfactory. The downlink power amplifier had adequate thermal conductivity at normal duty factors. At high duty factors it ran hot. The final installation added additional thermal epoxy and metal contacts to bring heat out to the shell of the satellite.

As final assembly was completed a second series of thermal-only tests were run. These tests were to verify workmanship through thermal cycling. One problem discovered was the need to carefully inspect motherboard-to-circuit board mating during assembly. This is difficult because of access, but assembly procedures were adapted.

3.8.2 Vibration Test

The principal acceptance test for SEDSAT was vibration. SEDSAT was shaken to protoflight levels, 3 dB above maximum flight levels. The tests were conducted twice, once in June of 1998 and again in August of 1998. The levels in August were acceptance (maximum flight only) to verify the board rework needed. The vibration test configuration used is shown in figure 3.8.2-1 and 3.8.2-2.

All vibration tests were conducted using a heavyweight PAF designed for the Sunsat/Oersted missions. This PAF was used because the test PAF for the standard installation (like SEDSAT) was not built to flight stiffness standards. It was considered essential that the PAF should be stiff, given that the flight PAF was also designed to be quite stiff. The larger PAF was identical in spacecraft mating to the lighterweight version, but had a different lower mounting arrangement.

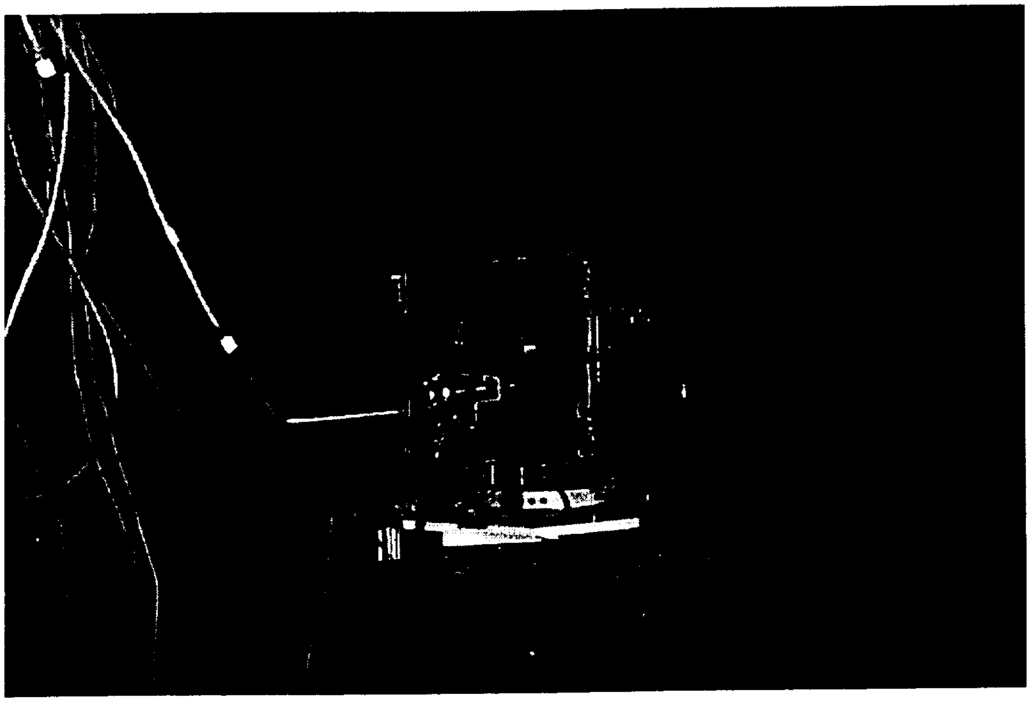


Figure 3.8.2-1 SEDSAT configured for the Z direction shake test.

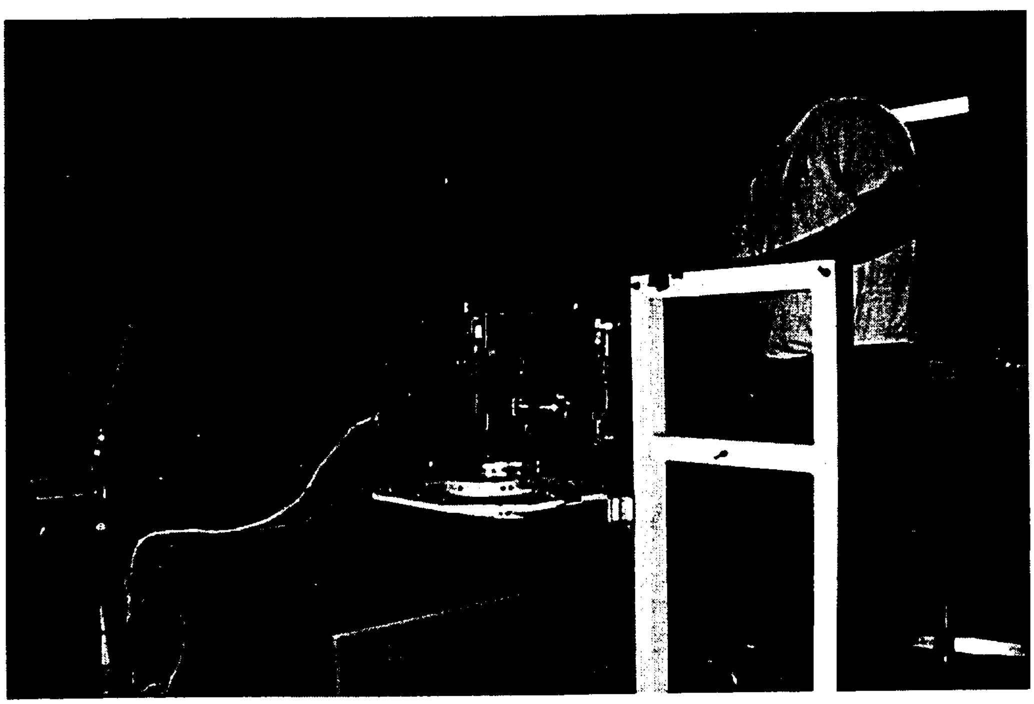


Figure 3.8.2-1 SEDSAT configured for the Z direction shake test.

The only unexpected result encountered in the vibration test was a substantial resonance at approximately 80 Hz in both the X and Y directions, but absent in the Z direction. It is possible this was a bending mode of SEDSAT, but the mode was not predicted by structural modeling. SEDSAT's assessment is that is most likely a rocking mode of the spacecraft on the PAF. This mode was not predicted in Delta II secondary data and should probably be further investigated.

3.8.3 Shock Test

The final acceptance test was a shock test implemented by releasing the PAF from the spacecraft using the flight pyrotechnics. For this test SEDSAT was elevated with the PAF attached and armed. The pyrotechnic cutters were fired and the PAF allowed to drop free. The configuration just before firing is shown in 3.8.3-1 and immediately after firing in 3.8.3-2.

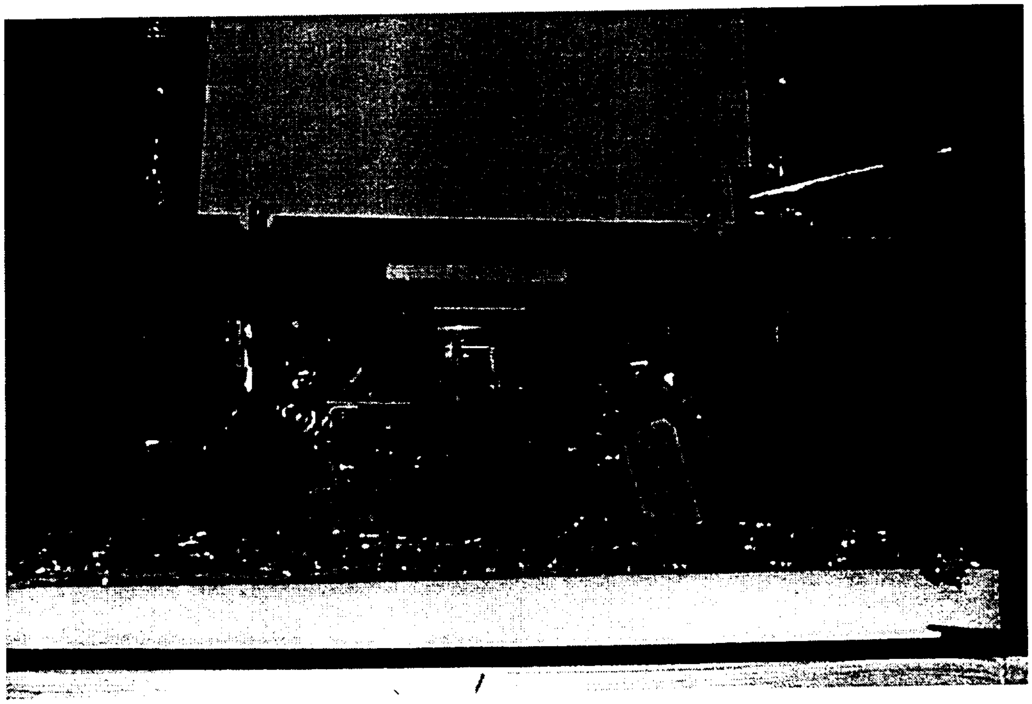


Figure 3.8.3-1 SEDSAT ready for shock test

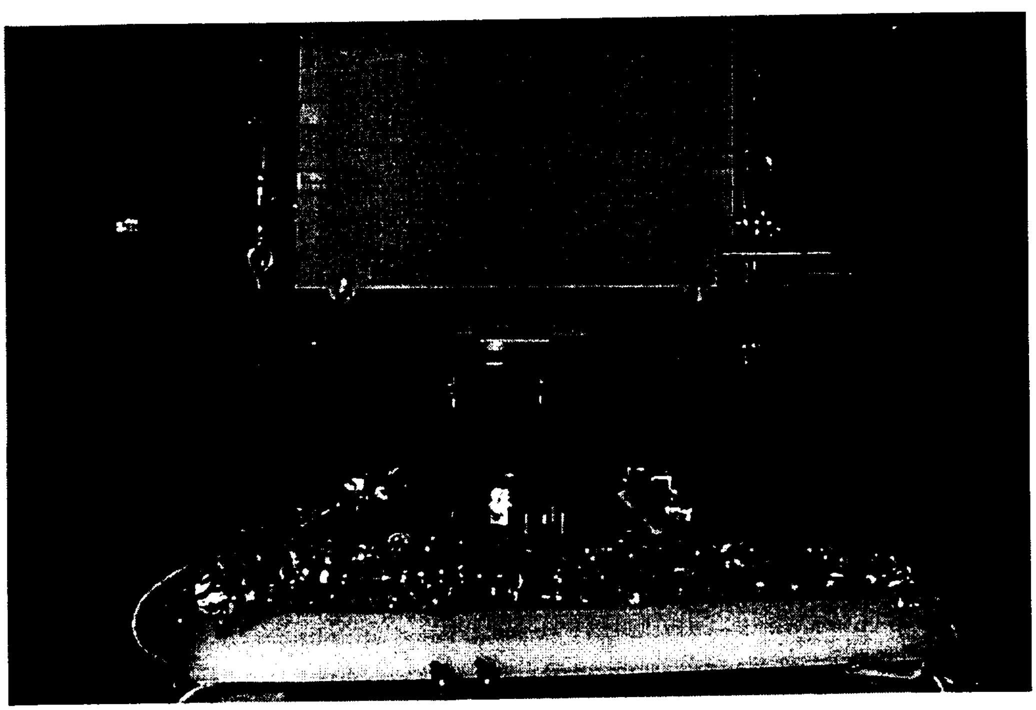


Figure 3.8.3-2 immediately after shock test

A serious problem occurred after the first shock test. Flight arming plugs were installed in SEDSAT to test the automatic startup. At the first shock test an incorrect plug was installed that shorted the batteries to ground. The resulting overcurrent through an internal connector burned printed circuit board coatings and insulating tape. Two sides of SEDSAT had to be removed, the motherboard replaced, and extensive cleaning done. After repair the vibration test was rerun at acceptance levels and the shock test repeated. This repeated tests were fully successful.

4 Launch Operations

SEDSAT was packed in its transport container, as shown in figure 4-1, and delivered to the Cape Canaveral launch facilities by automobile.

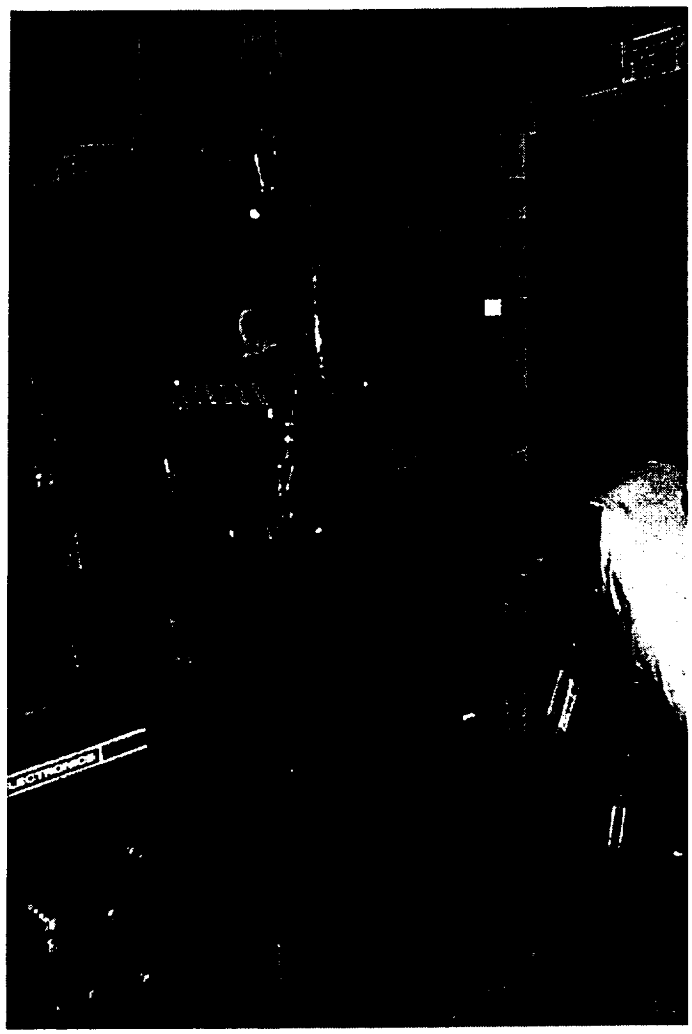
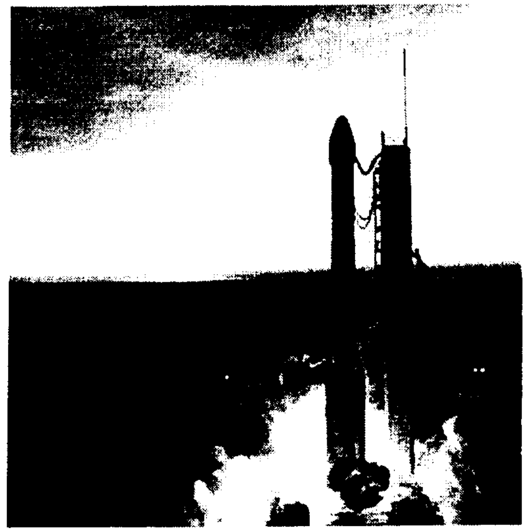


Figure 4-1 SEDSAT being packed for delivery

SEDSAT was launched on Oct. 25, figure 4-2. It separated as planned at 5003 seconds. Telemetry was detected almost immediately by the University of Arizona team, although it required three passes to begin to decode the telemetry correctly.



5 Orbit Data

The Norad two line elements for SEDSAT and the Delta II second stage are:

SEDSAT-1

1 25509U 98061B 98313.40161651 .00001107 00000-0 22293-3 0 73

2 25509 31.4468 305.9875 0369457 27.6957 334.2835 14.23748531 2262

DELTA 2 R/B 1

1 25510U 98061C 98313.25271565 .00001836 00000-0 36397-3 0 488

2 25510 31.4473 306.7071 0373038 26.5041 335.4155 14.24541995 2241

This represents a nearly nominal orbit.

The early health of the spacecraft can easily be seen in the orbit 14 telemetry collected at the University of Arizona, attached as Table 5-1. This shows SEDSAT operating in the dark and then breaking into the sunlight part of the way through the pass. During the eclipse the system is operating nominally at approximately 15 watts power consumption. As it breaks out of eclipse the solar panels immediately begin generating power. The last column shows the calculated power production of the panels. This number is lower than expected. Partially this is because the panels are cold having just emerged from an eclipse. The variability in power from the panels is indicative of the slow spin the satellite is in.

The telemetry cannot indicate the source of the uplink problem. After many attempts at uplinking without any evidence that any data was being received at all we have concluded that something in the uplink receive path broke somewhere between delivery and orbit. There are several reasonable places at which a break could have taken place. Among them are the antenna on the –Z face plate, the antenna-cable attachment, the cable-LNA connection, solder joints on the Mode-L receiver, and solder joints or parts around the CDS serial port chips. With the data available there is no way to be certain.

It is possible that there is some other source of the problem, but no scenario is convincing. Absent an uplink SEDSAT will send telemetry for the indefinite future and will allow the battery and electronics experiments to continue. During unfavorable precession periods the satellite will have a net negative power budget and will fully discharge its batteries. When this occurs, and it has already been observed, the spacecraft will reactivate at the end of an eclipse. It can detect that it went into a discharge reset and goes into a power conservative mode. As noted, this has already been observed and is expected to continue as long as the batteries hold a charge sufficient to last one orbit in the most power conservative mode.

NCC9-49

Date	10/25/98				Solar Panel Currents (mA)						
Tucson Time	Main Volts (V)	Main Current (mA)	Main Power (Watts)	Calc Charge (Ah)	+X:	+Y:	+Z:	-X:	-Y:	XYZ Total	Panel Power
4:20:39	19.32	0.78	15.03	5.43	0	0	0	0	0	0	0.00
4:20:59	19.29	0.78	15.12	5.43	0	0	0	0	0	0	0.00
4:21:19	19.32	0.78	15.15	5.42	0	0	0	0	0	0	0.00
4:21:39	19.32	0.78	15.03	5.42	0	0	0	0	0	0	0.00
4:21:59	19.29	0.78	15.12	5.42	0	0	0	0	0	0	0.00
4:22:19	19.29	0.78	15.12	5.42	0	0	0	0	0	0	0.00
4:22:39	19.32	0.78	15.11	5.42	0	0	0	0	0	0	0.00
4:22:59	19.29	0.79	15.18	5.41	0	0	0	0	0	0	0.00
4:23:19	19.29	0.78	15.12	5.41	0	0	0	0	0	0	0.00
4:23:39	19.29	0.78	15.09	5.41	0	0	0	0	0	0	0.00
4:23:59	19.29	0.79	15.18	5.41	0	0	0	0	0	0	0.00
4:24:19	19.29	0.78	15.12	5.40	0	0	0	0	0	0	0.00
4:24:59	19.29	0.78	15.09	5.40	0	0	0	0	0	0	0.00
4:25:19	19.29	0.78	15.12	5.40	0	0	0	0	0	0	0.00
4:25:39	19.29	0.78	15.12	5.40	0	0	0	0	0	0	0.00
4:25:59	19.29	0.78	15.09	5.39	0	0	0	0	0	0	0.00
4:26:19	19.29	0.78	15.12	5.39	0	0	0	0	0	0	0.00
4:26:39	19.29	0.79	15.18	5.39	0	0	0	0	0	0	0.00
4:26:59	19.29	0.78	15.12	5.39	0	0	0	0	0	0	0.00
4:27:19	19.29	0.78	15.12	5.39	0	0	0	0	0	0	0.00
4:27:39	19.29	0.78	15.12	5.38	0	0	0	0	0	0	0.00
4:28:00	19.26	0.78	15.10	5.38	0	0	0	0	0	0	0.00
4:28:20	19.29	0.78	15.12	5.38	0	0	0	0	0	0	0.00
4:28:40	19.29	0.78	15.12	5.38	0	0	0	0	0	0	0.00
4:29:00	19.45	0.78	15.09	5.38	0	227	0	429	302	958	18.64
4:29:20	19.45	0.77	15.06	5.37	0	0	195	375	366	936	18.21
4:29:40	19.45	0.77	15.06	5.37	0	0	185	449	0	634	12.33
4:30:00	19.36	0.78	15.14	5.37	0	161	0	0	0	161	3.12
4:30:19	19.48	0.77	15.08	5.37	0	161	0	0	0	161	3.14
4:30:20	19.48	0.77	15.08	5.37	512	395	0	0	0	907	17.67
4:32:19	19.48	0.77	15.08	5.37	0	0	227	0	612	839	16.35
4:32:39	19.55	0.77	15.07	5.37	483	0	471	0	0	954	18.65
4:32:59	19.48	0.78	15.24	5.37	65	0	185	0	0	250	4.87
4:33:59	19.52	0.77	15.10	5.37	0	0	109	532	239	880	17.17
4:34:19	19.48	0.78	15.12	5.37	0	0	0	419	0	419	8.16
4:34:39	19.42	0.78	15.11	5.37	61	236	0	0	0	297	5.77
4:34:59	19.55	0.77	15.13	5.37	507	417	0	0	78	1002	19.59
4:35:19	19.58	0.77	15.10	5.37	26	139	0	0	703	868	16.99
4:35:39	19.58	0.77	15.10	5.37	0	0	417	373	178	968	18.95
4:35:59	19.52	0.77	15.10	5.37	0	0	305	236	0	541	10.56
4:36:19	19.48	0.77	15.08	5.37	0	36	0	571	0	607	11.83
4:36:39	19.55	0.77	15.07	5.37	0	0	0	434	500	934	18.26
4:36:59	19.58	0.77	15.15	5.37	85	0	307	0	541	933	18.27
4:37:19	19.58	0.77	15.15	5.37	454	0	505	0	0	959	18.78

4:37:39	19.45	0.78	15.21	5.37	0	0	68	31	0	99	1.93
4:37:59	19.52	0.77	15.10	5.37	0	546	0	209	0	755	14.73
4:38:19	19.58	0.77	15.10	5.37	0	334	0	227	554	1115	21.83
4:38:39	19.55	0.77	15.07	5.37	0	0	26	617	148	791	15.46
4:38:59	19.52	0.78	15.18	5.37	0	0	0	373	0	373	7.28
4:39:20	19.48	0.78	15.12	5.37	146	305	0	0	0	451	8.79
4:39:40	19.58	0.77	15.15	5.37	446	473	0	0	205	1124	22.01